

NASA/SP—1998-7037/SUPPL388
November 27, 1998

AERONAUTICAL ENGINEERING

A CONTINUING BIBLIOGRAPHY WITH INDEXES



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Table of Contents

Records are arranged in categories 1 through 19, the first nine coming from the Aeronautics division of *STAR*, followed by the remaining division titles. Selecting a category will link you to the collection of records cited in this issue pertaining to that category.

01	Aeronautics	1
02	Aerodynamics	1
	Includes aerodynamics of bodies, combinations, wings, rotors, and control surfaces; and internal flow in ducts and turbomachinery.	
03	Air Transportation and Safety	35
	Includes passenger and cargo air transport operations; and aircraft accidents.	
04	Aircraft Communications and Navigation	N.A.
	Includes digital and voice communication with aircraft; air navigation systems (satellite and ground based); and air traffic control.	
05	Aircraft Design, Testing and Performance	38
	Includes aircraft simulation technology.	
06	Aircraft Instrumentation	50
	Includes cockpit and cabin display devices; and flight instruments.	
07	Aircraft Propulsion and Power	51
	Includes prime propulsion systems and systems components, e.g., gas turbine engines and compressors; and onboard auxiliary power plants for aircraft.	
08	Aircraft Stability and Control	55
	Includes aircraft handling qualities; piloting; flight controls; and autopilots.	
09	Research and Support Facilities (Air)	76
	Includes airports, hangars and runways; aircraft repair and overhaul facilities; wind tunnels; shock tubes; and aircraft engine test stands.	
10	Astronautics	77
	Includes astronautics (general); astrodynamics; ground support systems and facilities (space); launch vehicles and space vehicles; space transportation; space communications, spacecraft communications, command and tracking; spacecraft design, testing and performance; spacecraft instrumentation; and spacecraft propulsion and power.	
11	Chemistry and Materials	82
	Includes chemistry and materials (general); composite materials; inorganic and physical chemistry; metallic materials; nonmetallic materials; propellants and fuels; and materials processing.	

12	Engineering	83
	Includes engineering (general); communications and radar; electronics and electrical engineering; fluid mechanics and heat transfer; instrumentation and photography; lasers and masers; mechanical engineering; quality assurance and reliability; and structural mechanics.	
13	Geosciences	N.A.
	Includes geosciences (general); earth resources and remote sensing; energy production and conversion; environment pollution; geophysics; meteorology and climatology; and oceanography.	
14	Life Sciences	93
	Includes life sciences (general); aerospace medicine; behavioral sciences; man/system technology and life support; and space biology.	
15	Mathematical and Computer Sciences	N.A.
	Includes mathematical and computer sciences (general); computer operations and hardware; computer programming and software; computer systems; cybernetics; numerical analysis; statistics and probability; systems analysis; and theoretical mathematics.	
16	Physics	94
	Includes physics (general); acoustics; atomic and molecular physics; nuclear and high-energy; optics; plasma physics; solid-state physics; and thermodynamics and statistical physics.	
17	Social Sciences	N.A.
	Includes social sciences (general); administration and management; documentation and information science; economics and cost analysis; law, political science, and space policy; and urban technology and transportation.	
18	Space Sciences	N.A.
	Includes space sciences (general); astronomy; astrophysics; lunar and planetary exploration; solar physics; and space radiation.	
19	General	N.A.

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Subject Term Index	ST-1
Author Index	PA-1

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Typical Report Citation and Abstract

- ❶ **19970001126** NASA Langley Research Center, Hampton, VA USA
- ❷ **Water Tunnel Flow Visualization Study Through Poststall of 12 Novel Planform Shapes**
- ❸ Gatlin, Gregory M., NASA Langley Research Center, USA Neuhart, Dan H., Lockheed Engineering and Sciences Co., USA;
- ❹ Mar. 1996; 130p; In English
- ❺ Contract(s)/Grant(s): RTOP 505-68-70-04
- ❻ Report No(s): NASA-TM-4663; NAS 1.15:4663; L-17418; No Copyright; Avail: CASI; A07, Hardcopy; A02, Microfiche
- ❼

To determine the flow field characteristics of 12 planform geometries, a flow visualization investigation was conducted in the Langley 16- by 24-Inch Water Tunnel. Concepts studied included flat plate representations of diamond wings, twin bodies, double wings, cutout wing configurations, and serrated forebodies. The off-surface flow patterns were identified by injecting colored dyes from the model surface into the free-stream flow. These dyes generally were injected so that the localized vortical flow patterns were visualized. Photographs were obtained for angles of attack ranging from 10° to 50°, and all investigations were conducted at a test section speed of 0.25 ft per sec. Results from the investigation indicate that the formation of strong vortices on highly swept forebodies can improve poststall lift characteristics; however, the asymmetric bursting of these vortices could produce substantial control problems. A wing cutout was found to significantly alter the position of the forebody vortex on the wing by shifting the vortex inboard. Serrated forebodies were found to effectively generate multiple vortices over the configuration. Vortices from 65° swept forebody serrations tended to roll together, while vortices from 40° swept serrations were more effective in generating additional lift caused by their more independent nature.
- ❽ Author
- ❾ *Water Tunnel Tests; Flow Visualization; Flow Distribution; Free Flow; Planforms; Wing Profiles; Aerodynamic Configurations*

Key

1. Document ID Number; Corporate Source
2. Title
3. Author(s) and Affiliation(s)
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AERONAUTICAL ENGINEERING

A Continuing Bibliography (Suppl. 388)

NOVEMBER 27, 1998

01 AERONAUTICS

19980228296 NASA Langley Research Center, Hampton, VA USA

Aeronautical Engineering: A Continuing Bibliography with Indexes, Supplement 386

Oct. 30, 1998; 33p; In English

Report No.(s): NASA/SP-1998-7037/SUPPL386; NAS 1.21:7037/SUPPL386; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This supplemental issue of Aeronautical Engineering, A Continuing Bibliography with Indexes (NASA/SP-1998-7037) lists reports, articles, and other documents recently announced in the NASA STI Database. The coverage includes documents on the engineering and theoretical aspects of design, construction, evaluation, testing, operation, and performance of aircraft (including aircraft engines) and associated components, equipment, and systems. It also includes research and development in aerodynamics, aeronautics, and ground support equipment for aeronautical vehicles. Each entry in the publication consists of a standard bibliographic citation accompanied, in most cases, by an abstract.

CASI

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02 AERODYNAMICS

Includes aerodynamics of bodies, combinations, wings, rotors, and control surfaces; and internal flow in ducts and turbomachinery.

19980227840 NASA Langley Research Center, Hampton, VA USA

Pressure and Force Characteristics of Noncircular Cylinders as Affected by Reynolds Number with a Method Included for Determining the Potential Flow About Arbitrary Shapes

Polhamus, Edward C., NASA Langley Research Center, USA; Geller, Edward W., NASA Langley Research Center, USA; Grunwald, Kalman J., NASA Langley Research Center, USA; 1959; 44p; In English

Report No.(s): NASA-TR-R-46; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The low-speed pressure-distribution and force characteristics of several noncircular two-dimensional cylinders were measured in wind tunnel through a range of Reynolds numbers and flow incidences. A method of determining the potential-flow pressure distribution for arbitrary cross sections is described. Application of the data in predicting the spin characteristics of fuselages is briefly discussed.

Author

Flow Distribution; Pressure Distribution; Potential Flow; Predictions; Fuselages

19980227841 NASA Langley Research Center, Hampton, VA USA

A Systematic Kernel Function Procedure for Determining Aerodynamic Forces on Oscillating or Steady Finite Wings at Subsonic Speeds

Watkins, Charles E., NASA Langley Research Center, USA; Woolston, Donald S., NASA Langley Research Center, USA; Cunningham, Herbert J., NASA Langley Research Center, USA; 1959; 24p; In English

Report No.(s): NASA-TR-R-48; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Details are given of a numerical solution of the integral equation which relates oscillatory or steady lift and downwash distributions in subsonic flow. The procedure has been programmed for the IBM 704 electronic data processing machine and yields

the pressure distribution and some of its integrated properties for a given Mach number and frequency and for several modes of oscillation in from 3 to 4 minutes, results of several applications are presented.

Author

Aerodynamic Forces; Kernel Functions; Wings; Subsonic Speed; Downwash; Data Processing Equipment; Force Distribution

19980227843 NASA Langley Research Center, Hampton, VA USA

Effect of Afterbody Terminal Fairings on the Performance of a Pylon-Mounted Turbojet-Nacelle Model

Willis, Conrad M., NASA Langley Research Center, USA; Mercer, Charles E., NASA Langley Research Center, USA; Mar. 1960; 42p; In English

Report No.(s): NASA-TM-X-215; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation of the effect of afterbody terminal fairings on the performance of a pylon-mounted turbojet-nacelle model has been conducted in the Langley 16-foot transonic tunnel. A basic afterbody having a boattail angle of 16 deg was investigated with and without terminal fairings. The equivalent boattail angle, based on the cross-sectional area of the afterbody and terminal fairings, was 8 deg. Therefore, a simple body of revolution with a boattail angle of 8 deg was included for comparison. The tests were made at an angle of attack of 0 deg, Mach numbers of 0.80 to 1.05, jet total-pressure ratio of 1 to approximately 5, and an average Reynolds number per foot of 4.1×10^6 . A hydrogen peroxide jet simulator was used to supply the hot-jet exhaust. The results indicate that addition of terminal fairings to a 16 deg boattail afterbody increased the thrust-minus-drag coefficients and provided the lowest effective drag of the three configurations tested.

Author

Fairings; Afterbodies; Turbojet Engines; Nacelles; Aerodynamic Drag; Boattails; Angle of Attack; Aerodynamic Coefficients

19980227844 NASA Ames Research Center, Moffett Field, CA USA

Wind-Tunnel Investigation at Supersonic Speeds of the Lift, Drag, Static-Stability and Control Characteristics of a 0.03-Scale Model of the B-70 Airplane

Daugherty, James C., NASA Ames Research Center, USA; Green, Kendal H., NASA Ames Research Center, USA; Nov. 30, 1961; 127p; In English

Report No.(s): NASA-TM-SX-641; A-520; AF-AM-199; No Copyright; Avail: CASI; A07, Hardcopy; A02, Microfiche

A 0.03-scale model of the B-70 airplane was tested at Mach numbers of 2.5, 3.0, and 3.5, and at various Reynolds numbers from 3.0 to 8.0 million. Deflected canard and elevon effects were measured. A drag evaluation was made for the canard and vertical tail components. The effects of a 3 deg inboard cant of the vertical tails were measured as were the effects of modifications to the contour of the sides of the lower fuselage. In addition, the effects of twisted wing tips and of wing tip deflection were determined. Tests were conducted with different sizes of boundary-layer transition elements to allow determination of all-turbulent drag coefficients for the model at the test conditions. The data are presented without analysis.

Author

Aerodynamic Drag; Static Stability; Supersonic Speed; Boundary Layer Transition; Dynamic Control; Aerodynamic Coefficients; B-70 Aircraft; Fuselages; Scale Models

19980227856 NASA Langley Research Center, Hampton, VA USA

Effects at Mach Numbers of 1.61 and 2.01 of Camber and Twist on the Aerodynamic Characteristics of Three Swept Wings Having the Same Planform

Landrum, Emma Jean, NASA Langley Research Center, USA; Czarnecki, K. R., NASA Langley Research Center, USA; Aug. 1961; 50p; In English

Report No.(s): NASA-TN-D-929; L-1189; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been made at Mach numbers of 1.61 and 2.01 to determine the aerodynamic characteristics of three wings having a sweepback of 50 deg at the quarter-chord line, a taper ratio of 0.20, an NACA 65A005 thickness distribution, and an aspect ratio of 3.5. One wing was flat, one had at each spanwise station an $a = 0$ mean line modified to have a maximum height of 4-percent chord, and one had a linear variation of twist with 6 deg of washout at the tip. Tests were made with natural and fixed transition at Reynolds numbers ranging from 1.2×10^6 to 3.6×10^6 through an angle-of-attack range of -20 deg to 20 deg. When compared with the flat wing, the effect of the linear variation of twist with 6 deg of washout at the tip was to increase the lift-drag ratio when the leading edge was subsonic; but little increase in lift-drag ratio was obtained when the leading edge was supersonic. Pitching moment was increased and gave a positive trim point without greatly affecting the rate of change of pitch-

ing moment with lift coefficient. For the cambered wing the high minimum drag resulted in comparatively low lift-drag ratios. In addition, the pitching moments were decreased so that a negative trim point was obtained.

Author

Aerodynamic Characteristics; Swept Wings; Wing Planforms; Mach Number; Minimum Drag; Aerodynamic Coefficients; Lift Drag Ratio

19980227859 NASA Langley Research Center, Hampton, VA USA

Simple Formulas for Stagnation-Point Convective Heat Loads in Lunar Return

Grant, Frederick C., NASA Langley Research Center, USA; Jul. 1961; 18p; In English

Report No.(s): NASA-TN-D-890; L-1615; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Simple formulas are given for the stagnation-point convective heat loads in lunar return for two operational modes. The two modes of operation analyzed are typical of moderate heating and of nearly minimum heat loads, respectively. The values of the parameters in a simple two-parameter formula for the total-heat load are given in the lift-drag-ratio range of 0.2 to 1.0 and in the peak loading range of 2g to 10g. For vehicles having a lift-drag ratio near 0.5, which is typical of many proposed lunar return vehicles, the nominal mode had about 20 percent more absorption than the nearly minimum mode.

Author

Convective Heat Transfer; Heating; Stagnation Point

19980227862 Massachusetts Inst. of Tech., Cambridge, MA USA

Stall Propagation in a Cascade of Airfoils

Kriebel, Anthony R., Massachusetts Inst. of Tech., USA; Seidel, Barry S., Massachusetts Inst. of Tech., USA; Schwind, Richard G., Massachusetts Inst. of Tech., USA; 1960; 52p; In English

Report No.(s): NASA-TR-R-61; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An experimental investigation of stall propagation in a stationary circular cascade in which high speed schlieren and interferometer photography is used is described. This investigation suggests an analytical approach to the study of stall propagation which is valid only for an isolated blade row in an infinite flow field but which is not restricted to small unsteady perturbations or to an assumed simplified cascade geometry. Conditions necessary for the existence of the assumed type of stall cells are described and equations are derived for the velocity of stall cell propagation. The propagation velocities predicted for the theoretical potential flow model correlate with all the experimental values measured in an isolated rotor within 15 percent. Analysis of the flow model leads to the prediction of a tendency for the assumed type of stall cell to split with increasing incidence of the mean flow through the blade row. This tendency appears to correlate with the experimental observation of a trend for increasing numbers of cells in the rotor.

Author

Airfoils; Potential Flow; Propagation Velocity; Flow Distribution; Perturbation

19980227873 NASA Lewis Research Center, Cleveland, OH USA

Friction and Pressure Drag of Boundary-Layer Diverter Systems at Mach Number of 3.0

Stitt, Leonard E., NASA Lewis Research Center, USA; Anderson, Bernhard H., NASA Lewis Research Center, USA; Jan. 1960; 22p; In English

Report No.(s): NASA-TM-X-147; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An experimental investigation was performed at a Mach number of 3.0 to determine the friction and pressure drags of a pylon and a 20 deg- and a 40 deg-included-angle wedge diverter over a range of Reynolds number. The results indicated that the measured friction drag coefficients agreed reasonably with that predicted by flat-plate theory. The pressure drag coefficients of the 20 and 40 deg wedges agreed with those presented in the literature. The total drag coefficient of the pylon and the 20 deg wedge diverter was about 0.36, based on diverter frontal area, while the drag coefficient of the 40 deg wedge was about 0.47.

Author

Boundary Layers; Flat Plates; Diverters; Wedges; Pylons; Pressure Drag; Friction Drag; Supersonic Speed; Wind Tunnel Tests

19980227876 NASA Langley Research Center, Hampton, VA USA

Aerodynamic Characteristics at a Mach Number of 3.10 of Several Fourth-Stage Shapes of the Scout Research Vehicle

Jaquet, Byron M., NASA Langley Research Center, USA; Jun. 1961; 18p; In English

Report No.(s): NASA-TN-D-916; L-1578; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A wind-tunnel investigation was made at a Mach number of 3.10 (Reynolds number per foot of $16.3 \times 10^{(exp 6)}$ to $16.9 \times 10^{(exp 6)}$) to determine the aerodynamic characteristics of various modifications of the payload section of the fourth stage of the

Scout research vehicle. It was found that, for the combination of stages 3 and 4, increasing the size of the nose of the basic Scout to provide a cylindrical section of the same diameter as the third stage increased the normal-force slope by about 30 percent, the axial force by about 39 percent, and moved the center of pressure forward by about one fourth-stage base diameter. by reducing the diameter of the cylinder, at about one nose length behind the base of the enlarged nose frustum, to that of the basic Scout and thereafter retaining the shape of the basic Scout, the center of pressure was moved rearward by about one-half fourth-stage base diameter at the expense of an additional 19-percent increase in axial force. A spike-hemisphere configuration had the largest forces and moments and the most forward center-of-pressure location of the configurations considered. Except for the axial force and pitching-moment slope, the experimental trends or magnitudes could not be estimated with the desired accuracy by Newtonian or-slender body theory.

Author

Aerodynamic Characteristics; Cylindrical Bodies; Magnitude; Pitching Moments; Slender Bodies; Research Vehicles

19980227964 NASA Langley Research Center, Hampton, VA USA

Some Examples of the Applications of the Transonic and Supersonic Area Rules to the Prediction of Wave Drag

Nelson, Robert L., NASA Langley Research Center, USA; Welsh, Clement J., NASA Langley Research Center, USA; Sep. 1960; 50p; In English

Report No.(s): NASA-TN-D-446; L-1000; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The experimental wave drags of bodies and wing-body combinations over a wide range of Mach numbers are compared with the computed drags utilizing a 24-term Fourier series application of the supersonic area rule and with the results of equivalent-body tests. The results indicate that the equivalent-body technique provides a good method for predicting the wave drag of certain wing-body combinations at and below a Mach number of 1. At Mach numbers greater than 1, the equivalent-body wave drags can be misleading. The wave drags computed using the supersonic area rule are shown to be in best agreement with the experimental results for configurations employing the thinnest wings. The wave drags for the bodies of revolution presented in this report are predicted to a greater degree of accuracy by using the frontal projections of oblique areas than by using normal areas. A rapid method of computing wing area distributions and area-distribution slopes is given in an appendix.

Author

Body-Wing Configurations; Mach Number; Wings; Aerodynamic Characteristics; Aeronautical Engineering

19980227968 NASA Langley Research Center, Hampton, VA USA

A Hydrogen Peroxide Hot-Jet Simulator for Wind-Tunnel Tests of Turbojet-Exit Models

Runkel, Jack F., NASA Langley Research Center, USA; Swihart, John M., NASA Langley Research Center, USA; Feb. 1959; 38p; In English

Report No.(s): NASA-MEMO-1-10-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A turbojet-engine-exhaust simulator which utilizes a hydrogen peroxide gas generator has been developed for powered-model testing in wind tunnels with air exchange. Catalytic decomposition of concentrated hydrogen peroxide provides a convenient and easily controlled method of providing a hot jet with characteristics that correspond closely to the jet of a gas turbine engine. The problems associated with simulation of jet exhausts in a transonic wind tunnel which led to the selection of a liquid monopropellant are discussed. The operation of the jet simulator consisting of a thrust balance, gas generator, exit nozzle, and auxiliary control system is described. Static-test data obtained with convergent nozzles are presented and shown to be in good agreement with ideal calculated values.

Author

Wind Tunnel Tests; Hydrogen Peroxide; Jet Exhaust; Simulators; Turbojet Engines; Gas Generators; Wind Tunnel Models; Simulation; Convergent Nozzles; Air Flow

19980227977 NASA Ames Research Center, Moffett Field, CA USA

Radiative Heat Transfer During Atmosphere Entry at Parabolic Velocity

Yoshikawa, Kenneth K., NASA Ames Research Center, USA; Wick, Bradford H., NASA Ames Research Center, USA; Radiative Heat Transfer at Parabolic Entry Velocity; Nov. 1961; 18p; In English; Lifting Manned Hypervelocity and Reentry Vehicles, 11-14 Apr. 1961, Hampton, VA, USA

Report No.(s): NASA-TN-D-1074; A-573; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Stagnation point radiative heating rates for manned vehicles entering the earth's atmosphere at parabolic velocity are presented and compared with corresponding laminar convective heating rates. The calculations were made for both nonlifting and

lifting entry trajectories for vehicles of varying nose radius, weight-to-area ratio, and drag. It is concluded from the results presented that radiative heating will be important for the entry conditions considered.

Author

Atmospheric Entry; Convective Heat Transfer; Aerospace Vehicles; Radiative Heat Transfer; Lifting Bodies; Aerodynamic Heating

19980227986 NASA Langley Research Center, Hampton, VA USA

Aerodynamic Loading Characteristics Including Effects of Aeroelasticity of a Thin-Trapezoidal-Wing-Body Combination at Mach Number of 1.43

Kelly, Thomas C., NASA Langley Research Center, USA; Sep. 1959; 42p; In English

Report No.(s): NASA-TM-X-119; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Results have been obtained in the Langley 8-foot transonic pressure tunnel at a Mach number of 1.43 and at angles of attack from 0 deg to about 24 deg which indicate the static-aerodynamic-loads characteristics for a 2-percent-thick trapezoidal wing in combination with a body. Included are the effects of changing Reynolds number and of fixing boundary-layer transition. The results show that aerodynamic loading characteristics at a Mach number of 1.43 are similar to those reported in NACA RM L56Jl2a for the same configuration at a Mach number of 1.115. Reducing the Reynolds number resulted in reductions in the deflection of the wing and caused a slight increase in the relative loading over the outboard wing sections since the deflections were in a direction to unload the tip sections. Little or no effects were seen to result from fixing boundary-layer transition at a tunnel stagnation pressure of 1,950 pounds per square foot.

Author

Aerodynamic Loads; Static Aerodynamic Characteristics; Boundary Layer Transition; Aeroelasticity; Trapezoidal Wings

19980227987 NASA Ames Research Center, Moffett Field, CA USA

Turbulent Skin Friction at High Mach Numbers and Reynolds Numbers in Air and Helium

Matting, Fred W., NASA Ames Research Center, USA; Chapman, Dean R., NASA Ames Research Center, USA; Nyholm, Jack R., NASA Ames Research Center, USA; Thomas, Andrew G., NASA Ames Research Center, USA; 1961; 46p; In English

Report No.(s): NASA-TR-R-82; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Results are given of local skin-friction measurements in turbulent boundary layers over an equivalent air Mach number range from 0.2 to 9.9 and an over-all Reynolds number variation of $2 \times 10^{(exp 6)}$ to $100 \times 10^{(exp 6)}$. Direct force measurements were made by means of a floating element. Flows were two-dimensional over a smooth flat surface with essentially zero pressure gradient and with adiabatic conditions at the wall. Air and helium were used as working fluids. An equivalence parameter for comparing boundary layers in different working fluids is derived and the experimental verification of the parameter is demonstrated. experimental results are compared with the results obtained by several methods of calculating skin friction in the turbulent boundary layer.

Author

Turbulent Boundary Layer; Skin Friction; Pressure Gradients; Adiabatic Conditions

19980227988 NASA Langley Research Center, Hampton, VA USA

Aerodynamic Characteristics of a Four-Propeller Tilt-Wing VTOL Model with Twin Vertical Tails, Including Effects of Ground Proximity

Grunwald, Kalman J., NASA Langley Research Center, USA; Jun. 1961; 38p; In English

Report No.(s): NASA-TN-D-901; L-1491; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Results are presented of a wind-tunnel investigation of the aerodynamic stability, control, and performance characteristics of a model of a four-propeller tilt-wing VTOL airplane employing flaps and speed brakes through the transition speed range. The results indicate that the wing was stalled for steady level flight for all conditions of the investigation; however, the flapped configuration did produce a higher maximum lift. The effectiveness of the flap in delaying the stall in the present investigation was not as great as in some previous investigations because the flap used was smaller than that used previously. The wing stall resulted in an appreciable reduction of aileron effectiveness during the transition. Out of ground effect the low horizontal tail did not appear to be in an adverse flow field as had been expected and showed no erratic changes in effectiveness; however, in ground effect a large nose-down moment was experienced by the model. In general, the lateral aerodynamic data indicate that the configuration

is directionally stable and possesses positive dihedral effect throughout the transition, and the data show no signs of erratic flow at the vertical tails.

Author

Aerodynamic Characteristics; Wind Tunnel Tests; Aerodynamic Stability; Vertical Takeoff Aircraft; Flapping; Ground Effect (Aerodynamics); Lateral Stability; Propeller Slipstreams

19980227990 NASA Langley Research Center, Hampton, VA USA

Tests of Aerodynamically Heated Multiweb Wing Structures in a Free Jet at Mach Number 2: Five Aluminum-Alloy Models of 20-Inch Chord with 0.064-Inch-Thick Skin, 0.025-Inch-Thick Webs, and Various Chordwise Stiffening at 2 deg Angle of Attack

Trussell, Donald H., NASA Langley Research Center, USA; Thomson, Robert G., NASA Langley Research Center, USA; Jan. 1960; 36p; In English

Report No.(s): NASA-TM-X-186; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An experimental study was made on five 2024-T3 aluminum-alloy multiweb wing structures (MW-2-(4), MW-4-(3), mw-16, MW-17, and MW-18), at a Mach number of 2 and an angle of attack of 2 deg under simulated supersonic flight conditions. These models, of 20-inch chord and semi-span and 5-percent-thick circular-arc airfoil section, were identical except for the type and amount of chordwise stiffening. One model with no chordwise ribs between root and tip bulkhead fluttered and failed dynamically partway through its test. Another model with no chordwise ribs (and a thinner tip bulkhead) experienced a static bending type of failure while undergoing flutter. The three remaining models with one, two, or three chordwise ribs survived their tests. The test results indicate that the chordwise shear rigidity imparted to the models by the addition of even one chordwise rib precludes flutter and subsequent failure under the imposed test conditions. This paper presents temperature and strain data obtained from the tests and discusses the behavior of the models.

Author

Aluminum Alloys; Wings; Airfoil Profiles; Supersonic Speed; Aeroelasticity; Stiffening; Free Jets; Structural Analysis

19980227991 NASA Langley Research Center, Hampton, VA USA

Calculation of Aerodynamic Loading and Twist Characteristics of a Flexible Wing at Mach Numbers Approaching 1.0 and Comparison with Experiment

Mugler, John P., Jr., NASA Langley Research Center, USA; 1960; 22p; In English

Report No.(s): NASA-TR-R-58; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An iteration method is presented by which the detailed aerodynamic loading and twist characteristics of a flexible wing with known elastic properties may be calculated. The method is applicable at Mach numbers approaching 1.0 as well as at subsonic Mach numbers. Calculations were made for a wing-body combination; the wing was swept back 45 deg and had an aspect ratio of 4. Comparisons were made with experimental results at Mach numbers from 0.80 to 0.98.

Author

Aerodynamic Loads; Body-Wing Configurations; Elastic Properties; Iteration; Flexible Wings

19980227993 NASA Langley Research Center, Hampton, VA USA

Wind-Tunnel Tests of Seven Static-Pressure Probes at Transonic Speeds

Capone, Francis J., NASA Langley Research Center, USA; Nov. 1961; 34p; In English

Report No.(s): NASA-TN-D-947; L-1563; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Wind-tunnel tests have been conducted to determine the errors of 3 seven static-pressure probes mounted very close to the nose of a body of revolution simulating a missile forebody. The tests were conducted at Mach numbers from 0.80 to 1.08 and at angles of attack from -1.7 deg to 8.4 deg. The test Reynolds number per foot varied from 3.35×10^6 to 4.05×10^6 . For three 4-vane, gimbaled probes, the static-pressure errors remained constant throughout the test angle-of-attack range for all Mach numbers except 1.02. For two single-vane, self-rotating probes having two orifices at ± 37.5 deg. from the plane of symmetry on the lower surface of the probe body, the static-pressure error varied as much as 1.5 percent of free-stream static pressure through the test angle-of- attack range for all Mach numbers. For two fixed, cone-cylinder probes of short length and large diameter, the static-pressure error varied over the test angle-of-attack range at constant Mach numbers as much as 8 to 10 percent of free-stream static pressure.

Author

Pressure Sensors; Static Pressure; Transonic Speed; Wind Tunnel Tests; Free Flow

19980228000 NASA Langley Research Center, Hampton, VA USA

Aerodynamic Characteristics of a Canard and an Outboard-Tail Airplane Model at High Subsonic Speeds

Fournier, Paul G., NASA Langley Research Center, USA; Nov. 1961; 70p; In English

Report No.(s): NASA-TN-D-1002; L-1284; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An investigation has been made in the Langley high-speed 7- by 10-foot tunnel through a range of Mach numbers from 0.60 to 0.95 of the static longitudinal and lateral stability and control characteristics of a canard airplane configuration and an outboard-tail configuration. The canard model had a twisted wing with approximately 67 deg of sweepback and an aspect ratio of 2.91 and was tested with three trapezoidal canard surfaces having ratios of exposed area to wing area of 0.032, 0.076, and 0.121. The canard model had a single body-mounted vertical tail. The outboard-tail model had its horizontal- and vertical-tail surfaces mounted on slender bodies attached to the wing tips and located to the rear and outboard of the 67 deg sweepback wing of aspect ratio 1.00. The data, which are presented with limited analysis, provide information at high subsonic speeds on these two types of high-speed airplanes which have previously been tested at supersonic speeds and reported in NACA RM L58BO7 and NACA RM L58E20.

Author

Aerodynamic Characteristics; Canard Configurations; Aircraft Models; Static Stability; Subsonic Speed; Sweepback Wings

19980228025 NASA Langley Research Center, Hampton, VA USA

Basic Pressure Measurements at Transonic Speeds on a Thin 45 deg Sweepback Highly Tapered Wing With Systematic Spanwise Twist Variations

Mugler, John P., Jr., NASA Langley Research Center, USA; Jan. 1959; 94p; In English

Report No.(s): NASA-MEMO-12-28-58L; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

Pressure distributions obtained in the Langley 8-foot transonic pressure tunnel on a thin, highly tapered, twisted, 45 deg sweepback wing in combination with a body are presented. The wing has a linear span-wise twist variation from 0 deg at 10 percent of the semispan to 6 deg at the tip. The tip is at a lower angle of attack than the root. Tests were made at stagnation pressures of 1.0 and 0.5 atmosphere, at Mach numbers from 0.800 to 1.200, and at angles of attack from -4 to 12 deg.

Author

Pressure Measurement; Wind Tunnel Tests; Transonic Speed; Sweepback Wings; Stagnation Pressure; Twisted Wings; Aerodynamic Loads; Aerodynamic Configurations

19980228026 NASA Langley Research Center, Hampton, VA USA

Several Methods for Reducing the Drag of Transport Configurations at High Subsonic Speeds

Whitcomb, Richard T., NASA Langley Research Center, USA; Heath, Atwood R., Jr., NASA Langley Research Center, USA; Mar. 1959; 14p; In English

Report No.(s): NASA-MEMO-2-25-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Results of investigations of several promising methods for alleviating the drag rise of transport configurations at high subsonic speeds are reviewed briefly. The methods include a wing leading-edge extension, a fuselage addition, and additions on the wing. Also, results are presented for a complete, improved transport configuration which incorporates the fuselage and wing additions and show that the improved configuration could have considerably higher cruise speeds than do current designs.

Author

Drag Reduction; Subsonic Speed; Aerodynamic Configurations; Aircraft Design; Aerodynamic Drag

19980228027 NASA Langley Research Center, Hampton, VA USA

A Wind-Tunnel Investigation of Rotor Behavior Under Extreme Operating Conditions with a Description of Blade Oscillations Attributed to Pitch-Lag Coupling

McKee, John W., NASA Langley Research Center, USA; Naeseth, Rodger L., NASA Langley Research Center, USA; Jan. 1959; 46p; In English

Report No.(s): NASA-MEMO-1-7-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A wind-tunnel investigation was made to study the behavior of a model helicopter rotor under extreme operating conditions. A 1/8-scale model of the front rotor of a tandem helicopter was built and tested to obtaining blade motion and rotor aerodynamic characteristics for conditions that could be encountered in high-speed pullout maneuvers. The data are presented without analysis. A description is given in an appendix of blade oscillations that were experienced during the course of the investigation and of the part that blade pitch-lag coupling played in contributing to the oscillatory condition.

Author

Rotary Wings; Rotors; Wind Tunnel Stability Tests; Turbomachine Blades

19980228028 NASA Dryden Flight Research Center, Edwards, CA USA

A Summary of Flight-Determined Transonic Lift and Drag Characteristics of Several Research Airplane Configurations

Bellman, Donald R., NASA Dryden Flight Research Center, USA; Apr. 1959; 58p; In English

Report No.(s): NASA-MEMO-3-3-59H; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Flight-determined lift and drag data from transonic flights of seven research airplane configurations of widely varying characteristics are presented and compared with wind-tunnel and rocket-model data. The airplanes are the X-5 (590 wing sweep), XF-92A, YF-102 with cambered wing, YF-102 with symmetrical wing, D-558-ii, X-3, and X-LE. The effects of some of the basic configuration differences on the lift and drag characteristics are demonstrated. As indicated by transonic similarity laws, most of the configurations demonstrate a relationship between the transonic increase in zero-lift drag and the maximum cross-sectional area. No such relationship was found between the drag-rise Mach number and its normally related parameters. A comparison of flight and wind-tunnel data shows a generally reasonable agreement, but Reynolds number differences can cause considerable variations in the drag levels of the flight and wind-tunnel tests. Maximum lift-drag ratios vary widely in the subsonic region as would be expected from differences in aspect ratio and wing thickness ratio; however, the variations diminish as the Mach number is increased through the transonic region. The attainment of maximum lift-drag ratio in level flight by several of the airplanes was limited by engine performance, stability characteristics, and buffet boundaries.

Author

Data Acquisition; Flight Tests; Transonic Flight; Wind Tunnel Models; Stability; Lift Drag Ratio

19980228030 NASA Ames Research Center, Moffett Field, CA USA

Lift-Drag Ratios for an Arrow Wing With Bodies at Mach Number 3

Jorgensen, Leland H., NASA Ames Research Center, USA; May 1959; 34p; In English

Report No.(s): NASA-MEMO-4-27-59A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Force and moment characteristics, including lift-drag ratios, have been measured for bodies of circular and elliptic cross section alone and combined with a warped arrow wing. The test Mach number was 2.94, and the Reynolds number was 3.5×10^6 (based on wing mean aerodynamic chord). The experimental results show that for equal volume the use of an elliptical body can result in a noticeably higher maximum lift-drag ratio than that obtained through use of a circular body. Methods for estimating the aerodynamic characteristics have been assessed by comparing computed with experimental results. Because of good agreement of the predictions with experiment, maximum lift-drag ratios have been computed for the arrow wing in combination with bodies of various sizes. These calculations have shown that, for an efficient wing-body combination, little loss in maximum lift-drag ratio results from considerable extension of afterbody length. For example, for a wing-body configuration having a maximum lift-drag ratio of about 7.1, a loss in maximum lift-drag ratio of less than 0.2 results from a 40-percent increase in body volume by extension of afterbody length. It also appears that with body length fixed, maximum lift-drag ratio decreases almost linearly with increase in body diameter. For a wing-body combination employing a body of circular cross section, a decrease in maximum lift-drag ratio from about 9.1 for zero body diameter to about 4.6 for a body diameter of 13.5 percent of the body length was computed.

Author

Lift Drag Ratio; Lifting Bodies; Afterbodies; Aerodynamic Characteristics

19980228032 NASA Langley Research Center, Hampton, VA USA

An Investigation of a 0.05-Scale Model of the XSM-64A Navaho Missile and Booster, Part 1, Force Study at Mach Numbers from 1.77 to 3.51

Church, James D., NASA Langley Research Center, USA; Taylor, Nancy L., NASA Langley Research Center, USA; 1959; 314p; In English

Report No.(s): NASA-MEMO-5-30-59L; L-348; No Copyright; Avail: CASI; A14, Hardcopy; A03, Microfiche

An investigation has been conducted in the Langley Unitary Plan wind tunnel to determine the aerodynamic loads and the static longitudinal and lateral stability of a 0.05-scale model of the XSM-64A Navaho missile and booster and its various components. Tests were conducted through a Mach number range of 1.77 to 3.51 with a corresponding Reynolds number range of 2.4×10^6 to 2.9×10^6 . Results are presented for an angle-of-attack range of -8 deg to 4 deg for the missile-booster combination and -10 deg to 10 deg for the missile-alone configuration. Tests for both configurations were conducted through an angle-of-sideslip range of -8 deg to 8 deg. Also presented are some effects on the model characteristics of the deflection of various components including canard, tip aileron, vertical stabilizer, speed brakes, and booster pitch and yaw thrust chambers. The various

components on which loads were measured include the wing, tip aileron, rudder, booster, booster separating surface, booster fin, and booster yaw and pitch thrust chambers. These data are presented without analysis.

Author

Experimentation; Booster Rockets; Aerodynamic Loads; Longitudinal Stability; Lateral Stability; Navaho Missile; Scale Models

19980228033 NASA Ames Research Center, Moffett Field, CA USA

The Effects of Streamwise-Deflected Wing Tips on the Aerodynamic Characteristics of an Aspect Ratio-2 Triangular Wing, Body, and Tail Combination

Peterson, Victor L., NASA Ames Research Center, USA; May 1959; 38p; In English

Report No.(s): NASA-MEMO-5-18-59A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been conducted on a triangular wing and body combination to determine the effects on the aerodynamic characteristics resulting from deflecting portions of the wing near the tips 90° to the wing surface about streamwise hinge lines. Experimental data were obtained for Mach numbers of 0.70, 1.30, 1.70, and 2.22 and for angles of attack ranging from -5 deg to +18 deg at sideslip angles of 0 deg and 5 deg. The results showed that the aerodynamic center shift experienced by the triangular wing and body combination as the Mach number was increased from subsonic to supersonic could be reduced by about 40 percent by deflecting the outboard 4 percent of the total area of each wing panel. Deflection about the same hinge line of additional inboard surfaces consisting of 2 percent of the total area of each wing panel resulted in a further reduction of the aerodynamic center travel of 10 percent. The resulting reductions in the stability were accompanied by increases in the drag due to lift and, for the case of the configuration with all surfaces deflected, in the minimum drag. The combined effects of reduced stability and increased drag of the untrimmed configuration on the trimmed lift-drag ratios were estimated from an analysis of the cases in which the wing-body combination with or without tips deflected was assumed to be controlled by a canard. The configurations with deflected surfaces had higher trimmed lift-drag ratios than the model with undeflected surfaces at Mach numbers up to about 1.70. Deflecting either the outboard surfaces or all of the surfaces caused the directional stability to be increased by increments that were approximately constant with increasing angle of attack at each Mach number. The effective dihedral was decreased at all angles of attack and Mach numbers when the surfaces were deflected.

Author

Body-Wing Configurations; Aerodynamic Drag; Wing Tips; Directional Stability; Canard Configurations; Lift Drag Ratio; Aerodynamic Characteristics; Aerodynamic Balance

19980228036 NASA Langley Research Center, Hampton, VA USA

Review of Aircraft Altitude Errors Due to Static-Pressure Source and Description of Nose-Boom Installations for Aerodynamic Compensation of Error

Gracey, William, NASA Langley Research Center, USA; Ritchie, Virgil S., NASA Langley Research Center, USA; Jun. 1959; 12p; In English

Report No.(s): NASA-MEMO-5-10-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A brief review of airplane altitude errors due to typical pressure installations at the fuselage nose, the wing tip, and the vertical fins is presented. A static-pressure tube designed to compensate for the position errors of fuselage-nose installations in the subsonic speed range is described. This type of tube has an ogival nose shape with the static-pressure orifices located in the low-pressure region near the tip. The results of wind-tunnel tests of these compensated tubes at two distances ahead of a model of an aircraft showed the position errors to be compensated to within 1/2 percent of the static pressure through a Mach number range up to about 1.0. This accuracy of sensing free-stream static pressure was extended up to a Mach number of about 1.15 by use of an orifice arrangement for producing approximate free-stream pressures at supersonic speeds and induced pressures for compensation of error at subsonic speeds.

Author

Position Errors; Nose Tips; Supersonic Speed; Static Pressure; Low Pressure

19980228038 NASA Lewis Research Center, Cleveland, OH USA

Pressure Drag of Axisymmetric Cowls Having Large Initial Lip Angles at Mach Numbers from 1.90 to 4.90

Samanich, Nick E., NASA Lewis Research Center, USA; Jan. 1959; 18p; In English

Report No.(s): NASA-MEMO-1-10-59E; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The results of experimental and theoretical data on nine cowls are presented to determine the effect of initial lip angle and projected frontal area on the cowl pressure drag coefficient at Mach numbers from 1.90 to 4.90. The experimental drag coefficients were approximated well with two-dimensional shock-expansion theory at the lower cowl-projected areas, but the difference between theory and experiment increased as the cowl area ratio was increased or as shock detachment at the cowl lips was

approached. An empirical chart is presented, which can be used to estimate the cowl pressure drag coefficient of cowls approaching an elliptic contour.

Author

Pressure Drag; Aerodynamic Drag; Aerodynamic Coefficients; Charts

19980228045 National Bureau of Standards, Washington, DC USA

Aerodynamic Heating and Fatigue

Kroll, Wilhelmina D., National Bureau of Standards, USA; Jun. 1959; 34p; In English

Report No.(s): NASA-MEMO-6-4-59W; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A review of the physical condition's under which future airplanes will operate has been made and the necessity for considering fatigue in the design has been established. A survey of the literature shows what phases of elevated-temperature fatigue have been investigated. Other studies that would yield data of particular interest to the designer of aircraft structures are indicated.

Author

Aerodynamic Heating; Aircraft Structures; Thermal Fatigue; High Temperature Tests

19980228048 NASA Langley Research Center, Hampton, VA USA

Aerodynamic Characteristics at Mach Numbers of 1.41 and 2.01 of a Series of Cranked Wings Ranging in Aspect Ratio from 4.00 to 1.74 in Combination with a Body

Sevier, John R., Jr., NASA Langley Research Center, USA; Jan. 1960; 42p; In English

Report No.(s): NASA-TM-X-172; L-261; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A program has been conducted in the Langley 4- by 4-foot supersonic pressure tunnel to determine the effects of certain wing plan-form variations on the aerodynamic characteristics of wing-body combinations at supersonic speeds. The present report deals with the results of tests of a family of cranked wing plan forms in combination with an ogive-cylinder body of revolution. Tests were made at Mach numbers of 1.41 and 2.01 at corresponding values of Reynolds number per foot of $3.0 \times 10(\exp 6)$ and $2.5 \times 10(\exp 6)$. Results of the tests indicate that the best overall characteristics were obtained with the low-aspect-ratio wings. Plan-form changes which involved decreasing the aspect ratio resulted in higher values of maximum lift-drag ratio, in addition to large increases in wing volume. Indications are that this trend would have continued to exist at aspect ratios even lower than the lowest considered in the present tests. Increases in the maximum lift-drag ratio of about 15 percent over the basic wing were achieved with practically no increase in drag. The severe longitudinal stability associated with the basic cranked wing was no longer present (within the limits of the present tests) on the wings of lower aspect ratio formed by sweeping forward the inboard portion of the trailing edge.

Author

Aerodynamic Characteristics; Lift Drag Ratio; Low Aspect Ratio Wings; Reynolds Number; Supersonic Speed; Swept Wings; Trailing Edges; Wing Planforms; Mach Number

19980228056 NASA Ames Research Center, Moffett Field, CA USA

Large-Scale Wind-Tunnel Tests of an Airplane Model with an Unswept, Aspect-Ratio-10 Wing, Two Propellers, and Blowing Flaps

Griffin, Roy N., Jr., NASA Ames Research Center, USA; Holzhauser, Curt A., NASA Ames Research Center, USA; Weiberg, James A., NASA Ames Research Center, USA; Dec. 1958; 52p; In English

Report No.(s): NASA-MEMO-12-3-58A; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An investigation was made to determine the lifting effectiveness and flow requirements of blowing over the trailing-edge flaps and ailerons on a large-scale model of a twin-engine, propeller-driven airplane having a high-aspect-ratio, thick, straight wing. With sufficient blowing jet momentum to prevent flow separation on the flap, the lift increment increased for flap deflections up to 80 deg (the maximum tested). This lift increment also increased with increasing propeller thrust coefficient. The blowing jet momentum coefficient required for attached flow on the flaps was not significantly affected by thrust coefficient, angle of attack, or blowing nozzle height.

Author

Boundary Layer Separation; Trailing Edge Flaps; Wind Tunnel Tests; High Aspect Ratio; Externally Blown Flaps; Lift; Boundary Layer Control; Blowing; Flapping; Jet Flow; Rectangular Wings; Separated Flow; Unswept Wings

19980228057 NASA Langley Research Center, Hampton, VA USA

Aerodynamic Characteristics of Towed Cones Used as Decelerators at Mach Numbers from 1.57 to 4.65

Charczenko, Nickolai, NASA Langley Research Center, USA; McShera, John T., NASA Langley Research Center, USA; Dec. 1961; 28p; In English

Report No.(s): NASA-TN-D-994; L-1505; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Towed and sting-supported cones were tested in the wake of various payloads at supersonic speeds to determine their drag and stability characteristics. The investigation extended over a Mach number range from 1.57 to 4.65 and included such variables as Reynolds number, cone angle, ratio of cone base diameter to payload base diameter, and trailing distance. The results of this investigation showed that the cones towed in the wake of a symmetrical payload at supersonic speeds, in general, have good drag and stability characteristics if towed in the supersonic flow region. A cone with an included angle between 80 deg and 90 deg will give maximum drag while still maintaining stability in the Mach number region of this investigation. In order to minimize wake effects, the ratio of cone base diameter to payload base diameter should be at least one and preferably around three. A trailing distance of three times the payload base diameter, in most cases, is of sufficient length to avoid low drag and instability of the decelerator.

Author

Aerodynamic Characteristics; Brakes (For Arresting Motion); Supersonic Flow; Payloads; Drag

19980228061 NASA Langley Research Center, Hampton, VA USA

Transonic Wind-Tunnel Tests of an Error-Compensated Static-Pressure Probe

Capone, Francis J., NASA Langley Research Center, USA; Aug. 1961; 17p; In English

Report No.(s): NASA-TN-D-961; L-1562; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation of the pressure-sensing characteristics of an error-compensated static-pressure probe mounted on the nose section of a missile body has been conducted in the Langley 16-foot transonic tunnel. The probe was free to rotate about its roll axis and was equipped with a vane so that the crossflow velocity component due to angles of attack or sideslip was always aligned with the probe's vertical plane of symmetry. The probe was tested in five axial positions with respect to the missile nose at Mach numbers from 0.30 to 1.08 and at angles of attack from -2.7 to 15.3 deg. The test Reynolds number per foot varied from 1.79×10^6 to 4.05×10^6 . Results showed that at a Mach number of 1.00 the static-pressure error decreased from 3.5 percent to 0.8 percent of the free-stream static pressure, as a result of a change in orifice location from 0.15 maximum missile diameter to 0.20 maximum missile diameter forward of the missile nose. Although compensation for pressure-sensing errors due to angles of attack up to 15.3 was maintained at Mach numbers from $M = 0.30$ to $M = 0.50$, there was an increase in error with an increase in angle of attack for Mach numbers between $M = 0.50$ and $M = 1.08$.

Author

Pressure Sensors; Wind Tunnel Tests; Cross Flow; Free Flow; Missile Bodies; Static Pressure; Errors

19980228062 NASA Ames Research Center, Moffett Field, CA USA

Atmosphere Entries with Vehicle Lift-Drag Ratio Modulated to Limit Deceleration and Rate of Deceleration: Vehicles with Maximum Lift-Drag Ratio of 0.5

Katzen, Elliott D., NASA Ames Research Center, USA; Levy, Lionel L., Jr., NASA Ames Research Center, USA; Dec. 1961; 38p; In English

Report No.(s): NASA-TN-D-1145; A-564; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An analysis has been made of atmosphere entries for which the vehicle lift-drag ratio was modulated to maintain specified maximum decelerations and/or maximum deceleration rates. The part of the vehicle drag polar used during modulation was from maximum lift coefficient to minimum drag coefficient. The entries were at parabolic velocity and the vehicle maximum lift-drag ratio was 0.5. Two-dimensional trajectory calculations were made for a nonrotating, spherical earth with an exponential atmosphere. The results of the analysis indicate that for a given initial flight-path angle, modulation generally resulted in a reduction of the maximum deceleration to 60 percent of the unmodulated value or a reduction of maximum deceleration rate to less than 50 percent of the unmodulated rate. These results were equivalent, for a maximum deceleration of 10 g, to lowering the undershoot boundary 24 miles with a resulting decrease in total convective heating to the stagnation point of 22 percent. However, the maximum convective heating rate was increased 18 percent; the maximum radiative heating rate and total radiative heating were each increased about 10 percent.

Author

Atmospheric Entry; Lift Drag Ratio; Aerodynamic Coefficients; Deceleration

19980228064 NASA Langley Research Center, Hampton, VA USA

Effect of Ground Proximity on the Aerodynamic Characteristics of Aspect-Ratio-1 Airfoils With and Without End Plates

Carter, Arthur W., NASA Langley Research Center, USA; Oct. 1961; 28p; In English

Report No.(s): NASA-TN-D-970; L-1693; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been made to determine the effect of ground proximity on the aerodynamic characteristics of aspect-ratio-1 airfoils. The investigation was made with the model moving over the water in a towing tank in order to eliminate the effects of wind-tunnel walls and of boundary layer on ground boards at small ground clearances. The results indicated that, as the ground was approached, the airfoils experienced an increase in lift-curve slope and a reduction in induced drag; thus, lift-drag ratio was increased. As the ground was approached, the profile drag remained essentially constant for each airfoil. Near the ground, the addition of end plates to the airfoil resulted in a large increase in lift-drag ratio. The lift characteristics of the airfoils indicated stability of height at positive angles of attack and instability of height at negative angles; therefore, the operating range of angles of attack would be limited to positive values. At positive angles of attack, the static longitudinal stability was increased as the height above the ground was reduced. Comparison of the experimental data with Wieselsberger's ground-effect theory (NACA Technical Memorandum 77) indicated generally good agreement between experiment and theory for the airfoils without end plates.

Author

Aerodynamic Characteristics; Ground Effect (Aerodynamics); Airfoils; End Plates; Aspect Ratio; Wind Tunnel Tests; Induced Drag

19980228066 NASA Langley Research Center, Hampton, VA USA

Free-Flight Aerodynamic-Heating Data to Mach Number 10.4 for a Modified Von Karman Nose Shape

Bland, William M., Jr., NASA Langley Research Center, USA; Collie, Katherine A., NASA Langley Research Center, USA; May 1961; 30p; In English

Report No.(s): NASA-TN-D-889; L-1610; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Aerodynamic-heating data have been obtained on a modified fineness-ratio-5.0 Von Karman nose shape at free-stream Mach numbers up to 10.4 with a rocket-propelled model. Transient skin temperatures were measured at one station, 26.6 inches behind the tip of a nose 31.6 inches long. A maximum skin temperature of 1,663 deg R was measured soon after the maximum Mach number was obtained. During the periods for which experimental Stanton numbers were presented, flow parameters just outside the boundary layer at the temperature measuring station varied as follows: the local Mach number varied in the range between 0.8 and 9.0 and the local Reynolds number varied in the range between $0.8 \times 10(\exp 6)$ and $35.5 \times 10(\exp 6)$. The ratio of skin temperature to local static temperature varied between 1.0 and 3.6. The experimental Stanton numbers agreed well with Van Driest's turbulent theory while the local Reynolds number was high - that is, while the local Reynolds number varied in a range above $6.8 \times 10(\exp 6)$. For local Reynolds numbers less than $3.5 \times 10(\exp 6)$ the experimental Stanton numbers were of the magnitude predicted by Van Driest's laminar theory. Transition from turbulent to laminar flow at the temperature measuring station, as indicated by the change in the magnitude of the Stanton number, occurred as the local Reynolds number decreased from $6.8 \times 10(\exp 6)$ to $3.5 \times 10(\exp 6)$ at essentially a constant local Mach number of about 9.0.

Author

Aerodynamic Heating; Flow Characteristics; Laminar Flow; Turbulent Flow; Free Flow; Fineness Ratio; Boundary Layers

19980228124 NASA, Washington, DC USA

Unsteady Motion of a Wing of Finite Span in a Compressible Medium

Krasilshchikova, E. A., NASA, USA; Izvestiia, Otdelenie Tekhnicheskikh Nauk; May 1961, No. 3, pp. 25-32; In English

Report No.(s): NASA-TT-F-58; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The linearized boundary value problem for the velocity potential for steady or unsteady finite wings in a compressible medium is formulated. The general equation for the velocity potential is given in the form of integrals of source distributions. The local source strength, determined in previous investigations (for example, in NACA TM-1383), is proportional to local downwash, but the region that must be covered by sources, as well as the domain of integration, must extend over all points of the xy-plane from which disturbances can affect the potential.

Author

Unsteady Flow; Wings; Linearity

19980228132 NASA Ames Research Center, Moffett Field, CA USA

Inlet Performance Characteristics from Wind-Tunnel Tests of a 0.10-Scale Air-Induction System Model of the YF-108A Airplane at Mach Numbers of 2.50, 2.76, and 3.00

Blackaby, James R., NASA Ames Research Center, USA; Lyman, E. Gene, NASA Ames Research Center, USA; Altermann, John A., III, NASA Ames Research Center, USA; Jul. 1959; 192p; In English

Report No.(s): NASA-MEMO-7-18-59A; AF-AM-157; No Copyright; Avail: CASI; A09, Hardcopy; A02, Microfiche

Inlet-performance and external-drag-coefficient characteristics are presented without analysis. Effects are shown of variations of fuselage boundary-layer diverter profile, bleed-surface porosity, bleed-exit area, and inlet ramp, and lip angle.

Author

Aerodynamic Coefficients; Wind Tunnel Tests; Scale Models; Engine Inlets; Diverters; Air Intakes; Aerodynamic Drag; Aerodynamic Configurations

19980228139 NASA Ames Research Center, Moffett Field, CA USA

Surface Pressure Distribution at Hypersonic Speeds for Blunt Delta Wings at Angle of Attack

Creager, Marcus O., NASA Ames Research Center, USA; May 1959; 20p; In English

Report No.(s): NASA-MEMO-5-12-59A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Surface pressures were measured over a blunt 60 deg delta wing with extended trailing edge at a Mach number of 5.7, a free-stream Reynolds number of 20,000 per inch, and angles of attack from -10 to +10 deg. Aft of four leading-edge thicknesses the pressure distributions evidenced no appreciable three-dimensional effects and were predicted qualitatively by a method described herein for calculation of pressure distribution in two-dimensional flow. Results of tests performed elsewhere on blunt triangular wings were found to substantiate the near two-dimensionality of the flow and were used to extend the range of applicability of the method of surface pressure predictions to Mach numbers of 11.5 in air and 13.3 in helium.

Author

Pressure Distribution; Delta Wings; Hypersonic Speed; Blunt Bodies; Free Flow; Two Dimensional Flow; Hypersonic Gliders; Aerodynamic Heating

19980228145 NASA Langley Research Center, Hampton, VA USA

Modified Matrix Method for Calculating Steady-State Span Loading on Flexible Wings in Subsonic Flight

Gainer, Patrick A., NASA Langley Research Center, USA; Aiken, William S., Jr., NASA Langley Research Center, USA; Jun. 1959; 64p; In English

Report No.(s): NASA-MEMO-5-26-59L; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

A method is presented for shortening the computations required to determine the steady-state span loading on flexible wings in subsonic flight. The method makes use of tables of downwash factors to find the necessary aerodynamic-influence coefficients for the application of lifting-line theory. Explicit matrix equations of equilibrium are converted into a matrix power series with a finite number of terms by utilizing certain characteristic properties of matrices. The number of terms in the series is determined by a trial-and-error process dependent upon the required accuracy of the solution. Spanwise distributions of angle of attack, airload, shear, bending moment, and pitching moment are readily obtained as functions of $q_m(\text{sub } R)$ where q denotes the dynamic pressure and mR denotes the lift-curve slope of a rigid wing. This method is intended primarily to make it practical to solve steady-state aeroelastic problems on the ordinary manually operated desk calculators, but the method is also readily adaptable to automatic computing equipment.

Author

Matrices (Mathematics); Matrix Methods; Steady State; Flexible Wings; Aerodynamic Loads; Subsonic Speed

19980228147 NASA Langley Research Center, Hampton, VA USA

Chordwise Pressure Distributions Over Several NACA 16-Series Airfoils at Transonic Mach Numbers up to 1.25

Ladson, Charles L., NASA Langley Research Center, USA; Jun. 1959; 290p; In English

Report No.(s): NASA-MEMO-6-1-59L; No Copyright; Avail: CASI; A13, Hardcopy; A03, Microfiche

A two-dimensional wind-tunnel investigation of the pressure distributions over several NACA 16-series airfoils with thicknesses of 4, 6, 9, and 12 percent of the chord and design lift coefficients of 0, 0.2, 1 and 0.5 has been conducted in the Langley airfoil test apparatus at transonic Mach numbers from 0.7 to 1.25. The tests ranged in Reynolds number from 2.4×10^6 to 2.8×10^6 and in angle of attack from -10 to 12 deg. Chordwise pressure distributions and schlieren flow photographs are presented without analysis.

Author

Airfoils; Pressure Distribution; Wind Tunnel Tests; Transonic Speed; Rotary Wings; Lift; Aerodynamic Coefficients

19980228156 NASA Langley Research Center, Hampton, VA USA

The Longitudinal Aerodynamic Characteristics of a Sweptback Wing-Body Combination With and Without End Plates at Mach Numbers from 0.40 to 0.93

Henderson, William P., NASA Langley Research Center, USA; May 1960; 22p; In English

Report No.(s): NASA-TN-D-389; L-834; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation was made at high subsonic speeds in the Langley high-speed 7- by 10-foot tunnel to determine the effect of end plates on the longitudinal aerodynamic characteristics of a sweptback wing-body combination with and without drooped chord-extensions. The wing had 45 deg sweepback of the quarter-chord line, an aspect ratio of 4, a taper ratio of 0.3, and NACA 65A006 airfoil sections parallel to the plane of symmetry, and was mounted near the rear of a body of revolution having a fineness ratio of approximately 8. The results indicated that the addition of the end plates to either the wing with drooped chord-extensions or to the wing without drooped chord-extensions slightly increased the lift in the low angle-of-attack range but slightly decreased the lift at moderate and high angles of attack. The addition of the end plates to the wing without the chord-extensions caused a small increase in the maximum lift-drag ratio at Mach numbers below 0.65 and a slight decrease at the higher Mach numbers; however, for the addition of the end plates to the wing with the chord-extensions the maximum lift-drag ratio was slightly decreased below a Mach number of 0.88, while a slight increase occurred for the higher Mach numbers. The addition of the end plates to the wings with and without the chord-extensions caused the static longitudinal stability to increase considerably for all Mach numbers; however, only a slight reduction in the aerodynamic-center variation with Mach number was observed.

Author

Body-Wing Configurations; Aerodynamic Characteristics; Airfoil Profiles; Sweepback; Lift Drag Ratio; Fineness Ratio; Afterbodies; Aerodynamic Balance

19980228161 NASA Langley Research Center, Hampton, VA USA

Effect of Groundboard Height on the Aerodynamic Characteristics of a Lifting Circular Cylinder Using Tangential Blowing from Surface Slots for Lift Generation

Lockwood, Vernard E., NASA Langley Research Center, USA; Oct. 1961; 26p; In English

Report No.(s): NASA-TN-D-969; L-1521; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A wind-tunnel investigation has been made to determine the ground effect on the aerodynamic characteristics of a lifting circular cylinder using tangential blowing from surface slots to generate high lift coefficients. The tests were made on a semispan model having a length 4 times the cylinder diameter and an end plate of 2.5 diameters. The tests were made at low speeds at a Reynolds number of approximately 290,000, over a range of momentum coefficients from 0.14 to 4.60, and over a range of groundboard heights from 1.5 to 10 cylinder diameters. The investigation showed an earlier stall angle and a large loss of lift coefficient as the groundboard was brought close to the cylinder when large lift coefficients were being generated. For example, at a momentum coefficient of 4.60 the maximum lift coefficient was reduced from a value of 20.3 at a groundboard height of 10 cylinder diameters to a value of 8.7 at a groundboard height of 1.5 cylinder diameters. In contrast to this there was little effect on the lift characteristics of changes in groundboard height when lift coefficients of about 4.5 were being generated. At a height of 1.5 cylinder diameters the drag coefficients generally increased rapidly when the slot position angle for maximum lift was exceeded. Slightly below the slot position angle for maximum lift, the groundboard had a beneficial effect, that is, the drag for a given lift was less near the groundboard than away from the groundboard. The variation of maximum circulation lift coefficient (maximum lift coefficient minus momentum coefficient) obtained in this investigation is in general agreement with a theory developed for a jet-flap wing which assumes that the loss in circulation is the result of blockage of the main stream beneath the wing.

Author

Aerodynamic Characteristics; Circular Cylinders; Ground Effect (Aerodynamics); Coefficients; Lift; Tangential Blowing

19980228180 Virginia Polytechnic Inst. and State Univ., Dept. of Engineering Science and Mechanics, Blacksburg, VA USA

Fluid, Structure, and Control Interaction Final Report, 1 May 1995 - 30 Apr. 1998

Inman, Daniel J.; Cattarius, Jens; Jun. 15, 1998; 7p; In English

Contract(s)/Grant(s): F49620-95-1-0362

Report No.(s): AD-A350942; AFRL-SR-BL-TR-98-0573; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

This report summarizes the activities in the use of linear piezoceramic linear actuators coupled with control methods and wing aeroelastic models to increase flutter speeds of wing/store configurations. The work resulted in the successful submission of three journal articles and the completion of three proceedings papers, one invited lecture, and the completion of one PhD dissertation. Our results examine the use of modern control methods (H infinity, loop transfer, LQG/LQR), applied to the wind/store flutter problem as modeled by a typical two-degree-of-freedom airfoil section coupled with the Jones' approximation to the Theodorsen function for the incompressible aerodynamic loads. Physical values are taken from the GBU-UB weapons system mounted on

an F-16. Actuator parameters are those of a piezoceramic, linear stroke, wafer device. Store aerodynamics are ignored. Results produce a substantial increase in flutter speed, and allow consideration of the store release problem. Preliminary results and modeling for solving the 3-D version were also investigated.

DTIC

Flutter; Actuators; Aeroelasticity; Piezoelectric Ceramics; Control Theory; Control Systems Design; Body-Wing Configurations

19980228192 NASA Langley Research Center, Hampton, VA USA

Some Divergence Characteristics of Low-Aspect-Ratio Wings at Transonic and Supersonic Speeds

Woolston, Donald S., NASA Langley Research Center, USA; Gibson, Frederick W., NASA Langley Research Center, USA; Cunningham, Herbert J., NASA Langley Research Center, USA; Sep. 1960; 46p; In English

Report No.(s): NASA-TN-D-461; L-582; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The problem of chordwise, or camber, divergence at transonic and supersonic speeds is treated with primary emphasis on slender delta wings having a cantilever support at the trailing edge. Experimental and analytical results are presented for four wing models having apex half-angles of 5 deg, 10 deg, 15 deg, and 20 deg. A Mach number range from 0.8 to 7.3 is covered. The analytical results include calculations based on small-aspect-ratio theory, lifting-surface theory, and strip theory. A closed-form solution of the equilibrium equation is given, which is based on low-aspect-ratio theory but which applies only to certain stiffness distributions. Also presented is an iterative procedure for use with other aerodynamic theories and with arbitrary stiffness distribution.

Author

Divergence; Low Aspect Ratio Wings; Supersonic Speed; Transonic Speed; Slender Wings; Models; Aspect Ratio

19980228200 NASA Langley Research Center, Hampton, VA USA

Wind-Tunnel Investigation of a Small-Scale Model of an Aerial Vehicle Supported by Tilting Ducted Fans

Smith, Charles C., Jr., NASA Langley Research Center, USA; Aug. 1960; 18p; In English

Report No.(s): NASA-TN-D-409; L-961; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A wind-tunnel investigation has been made to study the static longitudinal and lateral stability characteristics of a simplified aerial vehicle supported by ducted fans that tilt relative to the airframe. The ducts were in a triangular arrangement with one duct in front and two at the rear in order to minimize the influence of the downwash of the front duct on the rear ducts. The results of the investigation were compared with those of a similar investigation for a tandem two-duct arrangement in which the ducts were fixed (rather than tiltable) relative to the airframe, since the three-duct configuration had been devised in an attempt to avoid some of the deficiencies of the tandem fixed-duct configuration. The results of the investigation indicated that the tilting-duct arrangement had less noseup pitching moment for a given forward speed than the tandem fixed-duct arrangement. The model had less angle-of-attack instability than the tandem fixed-duct arrangement. The model was directionally unstable but had a positive dihedral effect throughout the test speed range.

Author

Wind Tunnel Tests; Scale Models; Longitudinal Stability; Airframes; Aerial Photography

19980228201 NASA Langley Research Center, Hampton, VA USA

Theoretical Calculations of the Pressures, Forces, and Moments Due to Various Lateral Motions Acting on Tapered Swept-back Vertical Tails with Supersonic Leading and Trailing Edges

Margolis, Kenneth, NASA Langley Research Center, USA; Elliott, Miriam H., NASA Langley Research Center, USA; Aug. 1960; 126p; In English

Report No.(s): NASA-TN-D-383; L-780; No Copyright; Avail: CASI; A07, Hardcopy; A02, Microfiche

Based on expressions for the linearized velocity potentials and pressure distributions given in NACA Technical Report 1268, formulas for the span load distribution, forces, and moments are derived for families of thin isolated vertical tails with arbitrary aspect ratio, taper ratio, and sweepback performing the motions constant sideslip, steady rolling, steady yawing, and constant lateral acceleration. The range of Mach number considered corresponds, in general, to the condition that the tail leading and trailing edges are supersonic. To supplement the analytical results, design-type charts are presented which enable rapid estimation of the forces and moments (expressed as stability derivatives) for given combinations of geometry parameters and Mach number.

Author

Numerical Analysis; Moments; Stabilizers (Fluid Dynamics); Pressure Distribution; Load Distribution (Forces); Aspect Ratio

19980228204 NASA Langley Research Center, Hampton, VA USA

Effect of Reynolds Number on the Force and Pressure Distribution Characteristics of a Two-Dimensional Lifting Circular Cylinder

Lockwood, Vernard E., NASA Langley Research Center, USA; McKinney, Linwood W., NASA Langley Research Center, USA; Sep. 1960; 28p; In English

Report No.(s): NASA-TN-D-455; L-936; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A two-dimensional lifting circular cylinder has been tested over a Mach number range from 0.011 to 0.32 and a Reynolds number range from 135,000 to 1,580,000 to determine the force and pressure distribution characteristics. Two flaps having chords of 0.37 and 6 percent of the cylinder diameter, respectively, and attached normal to the surface were used to generate lift. A third configuration which had 6-percent flaps 1800 apart was also investigated. All flaps were tested through a range of angular positions. The investigation also included tests of a plain cylinder without flaps. The lift coefficient showed a wide variation with Reynolds number for the 6-percent flap mounted on the bottom surface at the 50-percent-diameter station, varying from a low of about 0.2 at a Reynolds number of 165,000 to a high of 1.54 at a Reynolds number of 350,000 and then decreasing almost linearly to a value of 1.0 at a Reynolds number of 1,580,000. The pressure distribution showed that the loss of lift with Reynolds number above the critical was the result of the separation point moving forward on the upper surface. Pressure distributions on a plain cylinder also showed similar trends with respect to the separation point. The variation of drag coefficient with Reynolds number was in direct contrast to the lift coefficient with the minimum drag coefficient of 0.6 occurring at a Reynolds number of 360,000. At this point the lift-drag ratios were a maximum at a value of 2.54. Tests of a flap with a chord of 0.0037 diameter gave a lift coefficient of 0.85 at a Reynolds number of 520,000 with the same lift-drag ratio as the larger flap but the position of the flap for maximum lift was considerably farther forward than on the larger flap. Tests of two 6-percent flaps spaced 180 deg apart showed a change in the sign of the lift developed for angular positions of the flap greater than 132 deg at subcritical Reynolds numbers. These results may find use in application to aircraft using forebody strakes. The drag coefficient developed by the flaps when normal to the relative airstream was approximately equal to that developed by a flat plate in a similar attitude.

Derived from text

Reynolds Number; Pressure Distribution; Two Dimensional Models; Circular Cylinders; Aerodynamic Drag

19980228205 NASA Langley Research Center, Hampton, VA USA

Low-Speed Investigation of a Full-Span Internal-Flow Jet-Augmented Flap on a High-Wing Model with a 35 deg Swept Wing of Aspect Ratio 7.0

Turner, Thomas R., NASA Langley Research Center, USA; Aug. 1960; 42p; In English

Report No.(s): NASA-TN-D-434; L-931; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation of a full-span 17-percent-chord internal-flow jet-augmented flap on an aspect-ratio-7.0 wing with 35 deg of sweepback has been made in the Langley 300-MPH 7- by 10-foot tunnel. Blowing over the conventional elevator and blowing down from a nose jet were investigated as a means of trimming the large diving moments at the high momentum and high lift coefficients. The results of the investigation showed that the model with the horizontal tail 0.928 mean aerodynamic chord above the wing-chord plane was stable to the maximum lift coefficient. The large diving-moment coefficients could be trimmed either with a downward blowing nose jet or by blowing over the elevator. Neither the downward blowing nose jet nor blowing over the elevator greatly affected the static longitudinal stability of the model. Trimmed lift coefficients up to 8.8 with blowing over the elevator and up to 11.4 with blowing down at the nose were obtained when the flap was deflected 70 deg and the total momentum coefficients were 3.26 and 4.69.

Author

Low Speed; Experimentation; Aerodynamic Coefficients; Airfoil Profiles; Longitudinal Stability

19980228206 NASA Langley Research Center, Hampton, VA USA

Basic Pressure Measurements at Transonic Speeds on a Thin 45 deg Sweptback Highly Tapered Wing with Systematic Spanwise Twist Variations

Mugler, John P., Jr., NASA Langley Research Center, USA; Jun. 1959; 152p; In English

Report No.(s): NASA-MEMO-5-12-59L; No Copyright; Avail: CASI; A08, Hardcopy; A02, Microfiche

Pressure distributions obtained in the Langley 8-foot transonic pressure tunnel on a thin, highly tapered, twisted, 450 swept-back wing in combination with a body are presented. The wing has a cubic spanwise twist variation from 0 deg. at 10 percent of the semispan to 60 at the tip. The tip is at a lower angle of attack than the root. Tests were made at stagnation pressures of 1.0 and 0.5 atmosphere, at Mach numbers from 0.800 to 1.200, and at angles of attack from -4 deg. to 20 deg.

Author

Sweptback Wings; Swept Wings; Twisted Wings; Angle of Attack; Mach Number; Aerodynamics

19980228213 NASA Langley Research Center, Hampton, VA USA

Wind-Tunnel Investigation of Subsonic Longitudinal Aerodynamic Characteristics of a Tilttable-Wing Vertical-Take-Off-and-Landing Supersonic Bomber Configuration Including Turbojet Power Effects

Thompson, Robert F., NASA Langley Research Center, USA; Vogler, Raymond D., NASA Langley Research Center, USA; Moseley, William C., Jr., NASA Langley Research Center, USA; Jan. 1959; 92p; In English

Report No.(s): NASA-MEMO-1-8-59L; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

Jet-powered model tests were made to determine the low-speed longitudinal aerodynamic characteristics of a vertical-take-off and-landing supersonic bomber configuration. The configuration has a unique engine-wing arrangement wherein six large turbojet engines (three on each side of the fuselage) are buried in a low-aspect-ratio wing which is tilted into the vertical plane for take-off. An essentially two-dimensional variable inlet, spanning the leading edge of each wing semispan, provides air for the engines. Jet flow conditions were simulated for a range of military (nonafterburner) and afterburner turbojet-powered flight at subsonic speeds. Three horizontal tails were tested at a station down-stream of the jet exit and at three heights above the jet axes. A semi-span model was used and test parameters covered wing-fuselage incidence angles from 0 deg to 15 deg, wing angles of attack from -4 deg to 36 deg, a variable range of horizontal-tail incidence angles, and some variations in power simulation conditions. Results show that, with all horizontal tails tested, there were large variations in static stability throughout the lift range. When the wing and fuselage were aligned, the model was statically stable throughout the test range only with the largest tail tested (tail span of 1.25 wing span) and only when the tail was located in the low test position which placed the tail nearest to the undeflected jet. For transition flight conditions, none of the tail configurations provided satisfactory longitudinal stability or trim throughout the lift range. Jet flow was destabilizing for most of the test conditions, and varying the jet-exit flow conditions at a constant thrust coefficient had little effect on the stability of this model. Wing leading-edge simulation had some important effects on the longitudinal aerodynamic characteristics.

Author

Wind Tunnel Tests; Aerodynamic Characteristics; Subsonic Speed; Wings; Leading Edges; Vertical Landing; Takeoff; Bomber Aircraft; Turbojet Engines

19980228214 NASA Dryden Flight Research Center, Edwards, CA USA

Flow Characteristics About Two Thin Wings of Low Aspect Ratio Determined from Surface Pressure Measurements Obtained in Flight at Mach Numbers from 0.73 to 1.90

Taillon, Norman V., NASA Dryden Flight Research Center, USA; May 1959; 44p; In English

Report No.(s): NASA-MEMO-5-1-59H; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Surface pressure measurements were obtained at three chordwise stations on the wings of the X-3 and X-IE airplanes at Mach numbers from 0.73 to 1.13 for the X-3, and from 0.82 to 1.90 for the X-IE. Leading-edge separation is present on the X-3 wing at a Mach number of about 0.73 and an angle of attack of about 6 deg. However, when the Mach number is increased to 0.88, the trailing-edge separation dominates the pressure distribution and no leading-edge separation is visible although it is anticipated at the higher angles of attack shown. Conversely, the X-IE wing shows no indication of leading-edge separation within the scope of this investigation, but an overexpansion immediately behind the leading edge is present at a Mach number of approximately 0.82. Two separate normal shocks are present on the X-3 wing at a Mach number of about 0.88 and at a low angle of attack as an effect of wing geometry. These shocks merge to form a single shock when the angle of attack is increased to about 6 deg. At supersonic speeds the upper-surface expansion on the X-IE wing is limited by the approach of the pressure coefficients to the pressure coefficient for a vacuum.

Author

Pressure Measurement; X Wing Rotors; Thin Wings; Low Aspect Ratio; Flow Characteristics

19980228224 NASA Langley Research Center, Hampton, VA USA

Wind-Tunnel Investigation of the Effect of Angle of Attack and Flapping-Hinge Offset on Periodic Bending Moments and Flapping of a Small Rotor

McCarty, John Locke, NASA Langley Research Center, USA; Brooks, George W., NASA Langley Research Center, USA; Maglieri, Domenic J., NASA Langley Research Center, USA; Mar. 1959; 46p; In English

Report No.(s): NASA-MEMO-3-3-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A two-blade rotor having a diameter of 4 feet and a solidity of 0.037 was tested in the Langley 300-MPH 7- by 10-foot tunnel to obtain information on the effect of certain rotor variables on the blade periodic bending moments and flapping angles during the various stages of transformation between the helicopter and autogiro configuration. Variables studied included collective pitch angle, flapping-hinge offset, rotor angle of attack, and tip-speed ratio. The results show that the blade periodic bending moments generally increase with tip-speed ratio up into the transition region, diminish over a certain range of tip-speed ratio, and increase

again at higher tip-speed ratios. Above the transition region, the bending moments increase with collective pitch angle and rotor angle of attack. The absence of a flapping hinge results in a significant amplification of the periodic bending moments, the magnitudes of which increase with tip-speed ratio. When the flapping hinge is used, an increase in flapping-hinge offset results in reduced period bending moments. The aforementioned trends exhibited by the bending moments for changes in the variables are essentially duplicated by the periodic flapping motions. The existence of substantial amounts of blade stall increased both the periodic bending moments and the flapping angles. Harmonic analysis of the bending moments shows significant contributions of the higher harmonics, particularly in the transition region.

Derived from text

Wind Tunnel Tests; Angle of Attack; Flapping Hinges; Bending Moments; Rotors

19980228237 NASA Ames Research Center, Moffett Field, CA USA

A Numerical Method for Calculating the Wave Drag of a Configuration from the Second Derivative of the Area Distribution of a Series of Equivalent Bodies of Revolution

Levy, Lionel L., Jr., NASA Ames Research Center, USA; Yoshikawa, Kenneth K., NASA Ames Research Center, USA; Apr. 1959; 98p; In English

Report No.(s): NASA-MEMO-1-16-59A; No Copyright; Avail: CASI; A05, Hardcopy; A02, Microfiche

A method based on linearized and slender-body theories, which is easily adapted to electronic-machine computing equipment, is developed for calculating the zero-lift wave drag of single- and multiple-component configurations from a knowledge of the second derivative of the area distribution of a series of equivalent bodies of revolution. The accuracy and computational time required of the method to calculate zero-lift wave drag is evaluated relative to another numerical method which employs the Tchebichef form of harmonic analysis of the area distribution of a series of equivalent bodies of revolution. The results of the evaluation indicate that the total zero-lift wave drag of a multiple-component configuration can generally be calculated most accurately as the sum of the zero-lift wave drag of each component alone plus the zero-lift interference wave drag between all pairs of components. The accuracy and computational time required of both methods to calculate total zero-lift wave drag at supersonic Mach numbers is comparable for airplane-type configurations. For systems of bodies of revolution both methods yield similar results with comparable accuracy; however, the present method only requires up to 60 percent of the computing time required of the harmonic-analysis method for two bodies of revolution and less time for a larger number of bodies.

Author

Bodies of Revolution; Aerodynamic Configurations; Slender Bodies; Supersonic Speed; Wave Drag; Harmonic Analysis; Differential Equations

19980228239 NASA Langley Research Center, Hampton, VA USA

Wind-Tunnel Investigation of a Small-Scale Sweptback-Wing Jet-Transport Model Equipped with an External-Flow Jet-Augmented Double Slotted Flap

Johnson, Joseph L., Jr., NASA Langley Research Center, USA; Apr. 1959; 44p; In English

Report No.(s): NASA-MEMO-3-8-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A wind-tunnel investigation at low speeds has been made to study the aerodynamic characteristics of a small-scale sweptback-wing Jet-transport model equipped with an external-flow jet-augmented double slotted flap. Included in the investigation were tests of the wing alone to study the effects of varying the spanwise extent of blowing on the full-span flap. The results indicated that the double-slotted-flap arrangement of the present investigation was more efficient in terms of lift and drag than were the external-flow single-slotted-flap arrangements previously tested and gave a substantial reduction in the thrust-weight ratio required for a given lift coefficient under trimmed drag conditions. An increase in the spanwise extent of blowing on the full-span flap was also found to increase the efficiency of the model in terms of the lift and drag but, as would be expected on a sweptback-wing configuration, was accompanied by significant increases in negative pitching moment.

Author

Models; Jet Aircraft; Flapping; Aerodynamic Characteristics; Sweptback Wings

19980228240 NASA Langley Research Center, Hampton, VA USA

Wind-Tunnel Measurements of Effect of Dive-Recovery Flaps at Transonic Speeds on Models of a Seaplane and a Transport

Heath, Atwood R., Jr., NASA Langley Research Center, USA; Ward, Robert J., NASA Langley Research Center, USA; Jun. 1959; 52p; In English

Report No.(s): NASA-MEMO-6-9-59L; L-292; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

The effects of wing-lower-surface dive-recovery flaps on the aero- dynamic characteristics of a transonic seaplane model and a transonic transport model having 40 deg swept wings have been investigated in the Langley 16-foot transonic tunnel. The seaplane model had a wing with an aspect ratio of 5.26, a taper ratio of 0.333, and NACA 63A series airfoil sections streamwise. The transport model had a wing with an aspect ratio of 8, a taper ratio of 0.3, and NACA 65A series airfoil sections perpendicular to the quarter-chord line. The effects of flap deflection, flap longitudinal location, and flap sweep were generally investigated for both horizontal-tail-on and horizontal-tail-off configurations. Model force and moment measurements were made for model angles of attack from -5 deg to 14 deg in the Mach number range from 0.70 to 1.075 at Reynolds numbers of 2.95×10^6 to 4.35×10^6 . With proper longitudinal location, wing-lower-surface dive-recovery flaps produced lift and pitching-moment increments that increased with flap deflection. For the transport model a flap located aft on the wing proved to be more effective than one located more forward., both flaps having the same span and approximately the same deflection. For the seaplane model a high horizontal tail provided added effectiveness for the deflected-flap configuration.

Author

Wind Tunnel Tests; Flapping; Dynamic Characteristics; Models; Airfoil Profiles

19980228249 NASA Lewis Research Center, Cleveland, OH USA

Heat-Transfer and Friction Measurements with Variable Properties for Airflow Normal to Finned and Unfinned Tube Banks

Ragsdale, Robert G., NASA Lewis Research Center, USA; Dec. 1958; 40p; In English

Report No.(s): NASA-MEMO-10-9-58E; L-4880; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A single-line correlation of both the heat-transfer and pressure- drop data for electrically heated unfinned tubes is obtained by evaluating the density in the Reynolds number, specific heat, thermal conductivity, and viscosity at the film temperature, and the density in the friction coefficient at the bulk temperature. The heat-transfer data for finned tubes also exhibit an effect of physical-property variation which is removed by evaluating all properties, including density, at the primary surface temperature, and using $k^* = 0.015$ square root of $T/530$ for the thermal conductivity of air where T is the absolute temperature. The pressure drop for finned tubes is correlated by the use of bulk density in both the Reynolds number and friction coefficient. The data reported are for Reynolds numbers from 2000 to 35,000, surface temperatures from 600 to 1400 R, and an air inlet temperature of 530 R.

Author

Heat Transfer; Friction Measurement; Variability; Data Acquisition; Evaluation; Density (Mass/Volume)

19980228262 NASA Langley Research Center, Hampton, VA USA

Wind-Tunnel Investigation of Paraglider Models at Supersonic Speeds

Taylor, Robert T., NASA Langley Research Center, USA; Nov. 1961; 18p; In English

Report No.(s): NASA-TN-D-985; L-1490; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation was made in the Langley Unitary Plan wind tunnel to determine the behavior of paraglider models at moderate to high supersonic speeds. The models were deployed from a sting in the supersonic stream and steady-state aerodynamic performance data were obtained. Maximum values of the lift-drag ratio were about 1.4 at a Mach number of 2.65 and about 1.2 at a Mach number of 4.65. The angles of attack over which the models could be flown were limited by unsteady behavior of the canopy.

Author

Paragliders; Wind Tunnel Tests; Aerodynamic Characteristics; Lift Drag Ratio; Supersonic Flow; Supersonic Speed; Angle of Attack

19980228270 NASA Langley Research Center, Hampton, VA USA

Aerodynamic Characteristics of a Large-Scale Unswept Wing-Body-Tail Configuration with Blowing Applied Over the Flap and Wind Leading Edge

McLemore, H. Clyde, NASA Langley Research Center, USA; Peterson, John B., Jr., NASA Langley Research Center, USA; Sep. 1960; 214p; In English

Report No.(s): NASA-TN-D-407; L-927; No Copyright; Avail: CASI; A10, Hardcopy; A03, Microfiche

An investigation has been conducted in the Langley full-scale tunnel to determine the effects of a blowing boundary-layer-control lift-augmentation system on the aerodynamic characteristics of a large-scale model of a fighter-type airplane. The wing was unswept at the 70-percent- chord station, had an aspect ratio of 2.86, a taper ratio of 0.40, and 4-percent-thick biconvex airfoil sections parallel to the plane of symmetry. The tests were conducted over a range of angles of attack from approximately -4 deg to 23 deg for a Reynolds number of approximately 5.2×10^6 which corresponds to a Mach number of 0.08. Blowing rates were normally restricted to values just sufficient to control air-flow separation. The results of this investigation showed that wing

leading-edge blowing in combination with large values of wing leading-edge-flap deflection was a very effective leading-edge flow-control device for wings having highly loaded trailing-edge flaps. With leading-edge blowing there was no hysteresis of the lift, drag, and pitching-moment characteristics upon recovery from stall. End plates were found to improve the lift and drag characteristics of the test configuration in the moderate angle-of-attack range, and blockage to one-quarter of the blowing-slot area was not detrimental to the aerodynamic characteristics. Blowing boundary-layer control resulted in a considerably reduced landing speed and reduced landing and take-off distances. The ailerons were very effective lateral-control devices when used with blowing flaps.

Author

Aerodynamic Characteristics; Models; Wings; Wind Tunnel Tests; Airfoil Profiles; Body-Wing and Tail Configurations

19980228271 NASA Langley Research Center, Hampton, VA USA

Low-Speed Aerodynamic Characteristics of a Model of a Hypersonic Research Airplane at Angles of Attack up to 90 deg for a Range of Reynolds Numbers

Bowman, James S., Jr., NASA Langley Research Center, USA; Grantham, William D., NASA Langley Research Center, USA; Sep. 1960; 64p; In English

Report No.(s): NASA-TN-D-403; L-905; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Static force tests have been made at low subsonic speeds for a model of a hypersonic research airplane in the Langley high-speed 7- by 10-foot tunnel to determine the aerodynamic forces and moments up to an angle of attack of 90 deg for a range of Reynolds numbers. The Reynolds numbers, based on the mean aerodynamic chord, ranged from 740,000 to 1,900,000, which correspond to dynamic pressures from 15 to 100 lb/sq ft (Mach numbers from 0.10 to 0.27). The model was tested in the clean configuration with various horizontal-tail settings, horizontal tail off, lower rudder off, fuselage alone, and with various size strakes and slats on the nose of the model. Representative results of the present investigation are presented in plotted form, and a tabulation of all the data obtained is presented in a table. Appreciable effects on side force, yawing moment, and pitching moment are indicated by changes in Reynolds number for angles of attack of 40 to 90 deg.

Author

Aerodynamic Characteristics; Low Speed; Models; Hypersonics; Airships

19980228285 NASA Lewis Research Center, Cleveland, OH USA

Recovery Temperature, Transition, and Heat Transfer Measurements at Mach 5

Brinich, Paul F., NASA Lewis Research Center, USA; Aug. 1961; 52p; In English

Report No.(s): NASA-TN-D-1047; E-797; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Schlieren, recovery temperature, and heat-transfer measurements were made on a hollow cylinder and a cone with axes aligned parallel to the stream. Both the cone and cylinder were equipped with various bluntnesses, and the tests covered a Reynolds number range up to 20×10^6 at a free-stream Mach number of 4.95 and wall to free-stream temperature ratios from 1.8 to 5.2 (adiabatic). A substantial transition delay due to bluntness was found for both the cylinder and the cone. For the present tests (Mach 4.95), transition was delayed by a factor of 3 on the cylinder and about 2 on the cone, these delays being somewhat larger than those observed in earlier tests at Mach 3.1. Heat-transfer tests on the cylinder showed only slight effects of wall temperature level on transition location; this is to be contrasted to the large transition delays observed on conical-type bodies at low surface temperatures at Mach 3.1. The schlieren and the peak-recovery-temperature methods of detecting transition were compared with the heat-transfer results. The comparison showed that the first two methods identified a transition point which occurred just beyond the end of the laminar run as seen in the heat-transfer data.

Author

Heat Transfer; Transition Points; Experimentation; Temperature Measurement; Hypersonic Speed; Detection; Conical Bodies; Cylindrical Bodies

19980228287 NASA Langley Research Center, Hampton, VA USA

Transonic Aerodynamic Characteristics of Two Wedge Airfoil Sections Including Unsteady Flow Studies

Johnston, Patrick J., NASA Langley Research Center, USA; Jun. 1959; 48p; In English

Report No.(s): NASA-MEMO-4-30-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A two-dimensional wind-tunnel investigation has been conducted on a 20-percent-thick single-wedge airfoil section. Steady-state forces and moments were determined from pressure measurements at Mach numbers from 0.70 to about 1.25. Additional information on the flows about the single wedge is provided by means of instantaneous pressure measurements at Mach numbers up to unity. Pressure distributions were also obtained on a symmetrical double-wedge or diamond-shaped profile which had the same leading-edge included angle as the single-wedge airfoil. A comparison of the data on the two profiles to provide information

on the effects of the afterbody showed that with the exception of drag, the single-wedge profile proved to be aerodynamically superior to the diamond profile in all respects. The lift effectiveness of the single-wedge airfoil section far exceeded that of conventional thin airfoil sections over the speed range of the investigation. Pitching-moment irregularities, caused by negative loadings near the trailing edge, generally associated with conventional airfoils of equivalent thicknesses were not exhibited by the single-wedge profile. Moderately high pulsating pressures existing over the base of the single-wedge airfoil section were significantly reduced as the Mach number was increased beyond 0.92 and the boundaries of the dead airspace at the base of the model converged to eliminate the vortex street in the wake. Increasing the leading-edge radius from 0 to 1 percent of the chord had a minor effect on the steady-state forces and generally raised the level of pressure pulsations over the forward part of the single-wedge profile.

Author

Unsteady Flow; Aerodynamic Characteristics; Wedges; Airfoil Profiles; Leading Edges; Pressure Distribution; Thin Airfoils

19980228291 NASA Langley Research Center, Hampton, VA USA

Parasite-Drag Measurements of Five Helicopter Rotor Hubs

Churchill, Gary B., NASA Langley Research Center, USA; Harrington, Robert D., NASA Langley Research Center, USA; Feb. 1959; 28p; In English

Report No.(s): NASA-MEMO-1-31-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been conducted in the Langley full-scale tunnel to determine the parasite drag of five production-type helicopter rotor hubs. Some simple fairing arrangements were attempted in an effort to reduce the hub drag. The results indicate that, within the range of the tests, changes in angle of attack, hub rotational speed, and forward speed generally had only a small effect on the equivalent flat-plate area representing parasite drag. The drag coefficients of the basic hubs, based on projected hub frontal area, increased with hub area and varied from 0.5 to 0.76 for the hubs tested.

Author

Rotary Wings; Aerodynamic Drag; Hubs; Parasites; Aerodynamic Coefficients; Drag Measurement

19980228294 NASA Ames Research Center, Moffett Field, CA USA

Predicted Static Aeroelastic Effects on Wings with Supersonic Leading Edges and Streamwise Tips

Brown, Stuart C., NASA Ames Research Center, USA; Apr. 1959; 40p; In English

Report No.(s): NASA-MEMO-4-18-59A; A-159; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A method is presented for calculation of static aeroelastic effects on wings with supersonic leading edges and streamwise tips. Both chord-wise and spanwise deflections are taken into account. Aerodynamic and structural forces are introduced in influence coefficient form; the former are developed from linearized supersonic wing theory and the latter are assumed to be known from load-deflection tests or theory. The predicted effects of flexibility on lateral-control effectiveness, damping in roll, and lift-curve slope are shown for a low-aspect-ratio wing at Mach numbers of 1.25 and 2.60. The control effectiveness is shown for a trailing-edge aileron, a tip aileron, and a slot-deflector spoiler located along the 0.70 chord line. The calculations indicate that the tip aileron is particularly attractive from an aeroelastic standpoint, because the changes in effectiveness with dynamic pressure are small compared to the changes in effectiveness of the trailing-edge aileron and slot-deflector spoiler. The effects of making several simplifying assumptions in the example calculations are shown. The use of a modified strip theory to determine the aerodynamic influence coefficients gave adequate results only for the high Mach number case. Elimination of chordwise bending in the structural influence coefficients exaggerated the aeroelastic effects on rolling-moment and lift coefficients for both Mach numbers.

Author

Aeroelasticity; Supersonic Flow; Procedures; Computation; Static Aerodynamic Characteristics; Flexible Wings; Structural Influence Coefficients

19980228300 NASA Ames Research Center, Moffett Field, CA USA

Low-Speed Wind-Tunnel Investigation of Blowing Boundary-Layer Control on Leading- and Trailing-Edge Flaps of a Large-Scale, Low-Aspect-Ratio, 45 Swept-wing Airplane Configuration

Maki, Ralph L., NASA Ames Research Center, USA; Jan. 1959; 28p; In English

Report No.(s): NASA-MEMO-1-23-59A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Blowing boundary-layer control was applied to the leading- and trailing-edge flaps of a 45 deg sweptback-wing complete model in a full-scale low-speed wind-tunnel study. The principal purpose of the study was to determine the effects of leading-edge flap deflection and boundary-layer control on maximum lift and longitudinal stability. Leading-edge flap deflection alone was sufficient to maintain static longitudinal stability without trailing-edge flaps. However, leading-edge flap blowing was required to maintain longitudinal stability by delaying leading-edge flow separation when trailing-edge flaps were deflected either with or without blowing. Partial-span leading-edge flaps deflected 60 deg with moderate blowing gave the major increase in maximum

lift, although higher deflection and additional blowing gave some further increase. Inboard of 0.4 semispan leading-edge flap deflection could be reduced to 40 deg and/or blowing could be omitted with only small loss in maximum lift. Trailing-edge flap lift increments were increased by boundary-layer control for deflections greater than 45 deg. Maximum lift was not increased with deflected trailing-edge flaps with blowing.

Author

Wind Tunnel Tests; Trailing Edge Flaps; Leading Edge Flaps; Boundary Layer Control; Externally Blown Flaps; Boundary Layer Separation; Aircraft Control; Blowing; Sweptback Wings

19980228301 NASA Langley Research Center, Hampton, VA USA

Maximum Mean Lift Coefficient Characteristics at Low Tip Mach Numbers of a Hovering Helicopter Rotor Having an NACA 64(1)A012 Airfoil Section

Powell, Robert D., Jr., NASA Langley Research Center, USA; Feb. 1959; 28p; In English

Report No.(s): NASA-MEMO-1-23-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been conducted on the Langley helicopter test tower to determine experimentally the maximum mean lift-coefficient characteristics at low tip Mach number and a limited amount of drag-divergence data at high tip Mach number of a helicopter rotor having an NACA 64(1)A012 airfoil section and 8 deg of linear washout. Data are presented for blade tip Mach numbers $M(t)$ of 0.29 to 0.74 with corresponding values of tip Reynolds number of 2.59×10^6 and 6.58×10^6 . Comparisons are made between the data from the present rotor with results previously obtained from two other rotors: one having NACA 0012 airfoil sections and the other having an NACA 0009 airfoil tip section. At low tip Mach numbers, the maximum mean lift coefficient for the blade having the NACA 64(1)A012 section was about 0.08 less than that obtained with the blade having the NACA 0009 tip section and 0.21 less than the value obtained with the blade having the NACA 0012 tip section. Blade maximum mean lift coefficient values were not obtained for Mach number values greater than 0.47 because of a blade failure encountered during the tests. The effective mean lift-curve slope required for predicting rotor thrust varied from 5.8 for the tip Mach number range of 0.29 to 0.55 to a value of 6.65 for a tip Mach number of 0.71. The blade pitching-moment coefficients were small and relatively unaffected by changes in thrust coefficient and Mach number. In the instances in which stall was reached, the break in the blade pitching-moment curve was in a stable direction. The efficiency of the rotor decreased with an increase in tip speed. Expressed as figure of merit, at a tip Mach number of 0.29 the maximum value was about 0.74. Similar measurements made on another rotor having an NACA 0012 airfoil and with a rotor having an NACA 0009 tip section, showed a value of 0.75. Synthesized section lift and profile-drag characteristics for the rotor-blade airfoil section are presented as an aid in predicting the high-tip-speed performance of rotors having similar airfoils.

Author

Lift; Airfoil Profiles; Rotary Wings; Hovering; Aerodynamic Coefficients; Helicopters; Subsonic Speed

19980228310 NASA Ames Research Center, Moffett Field, CA USA

Subsonic Aerodynamic Characteristics of an Airplane Configuration with a 63 deg Sweptback Wing and Twin-Boom Tails

Savage, Howard F., NASA Ames Research Center, USA; Edwards, George G., NASA Ames Research Center, USA; Mar. 1959; 60p; In English

Report No.(s): NASA-MEMO-3-3-59A; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

A wind-tunnel investigation has been conducted to determine the effects of an unconventional tail arrangement on the subsonic static longitudinal and lateral stability characteristics of a model having a 63 deg sweptback wing of aspect ratio 3.5 and a fuselage. Tail booms, extending rearward from approximately the midsemispan of each wing panel, supported independent tail assemblies well outboard of the usual position at the rear of the fuselage. The horizontal-tail surfaces had the leading edge swept back 45 deg and an aspect ratio of 2.4. The vertical tail surfaces were geometrically similar to one panel of the horizontal tail. For comparative purposes, the wing-body combination was also tested with conventional fuselage-mounted tail surfaces. The wind-tunnel tests were conducted at Mach numbers from 0.25 to 0.95 with a Reynolds number of 2,000,000, at a Mach number of 0.46 with a Reynolds number of 3,500,000, and at a Mach number of 0.20 with a Reynolds number of 7,000,000. The results of the investigation indicate that longitudinal stability existed to considerably higher lift coefficients for the outboard tail configuration than for the configuration with conventional tail. Wing fences were necessary with both configurations for the elimination of sudden changes in longitudinal stability at lift coefficients between 0.3 and 0.5. Sideslip angles up to 15 deg had only small effects upon the pitching-moment characteristics of the outboard tail configuration. There was an increase in the directional stability for the outboard tail configuration at the higher angles of attack as opposed to a decrease for the conventional tail configuration at most of the Mach numbers and Reynolds numbers of this investigation. The dihedral effect increased rapidly with increasing angle

of attack for both the outboard and the conventional tail configurations but the increase was greater for the outboard tail configuration. The data indicate that the outboard tail is an effective roll control.

Author

Aerodynamic Characteristics; Body-Wing Configurations; Sweptback Wings; Directional Stability; Lateral Stability; Leading Edges

19980228314 NASA Ames Research Center, Moffett Field, CA USA

Experimental Wind-Tunnel Investigation of the Transonic Damping-in-Pitch Characteristics of Two Wing-Body Combinations

Emerson, Horace F., NASA Ames Research Center, USA; Robinson, Robert C., NASA Ames Research Center, USA; Dec. 1958; 30p; In English

Report No.(s): NASA-MEMO-11-30-58A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The results of an experimental wind-tunnel investigation of the damping in pitch of two wing-body combinations are presented. The tests were conducted in the Ames 14-foot transonic wind tunnel over a Mach number range from 0.60 to 1.18. Reynolds numbers varied from 2.3 million to 5.5 million. One model with a triangular wing of aspect ratio 2 having NACA 0003-63 sections was oscillated at an amplitude of 1.5 and a frequency of 17 cycles per second. The second model with a straight, tapered wing of aspect ratio 3 having 3-percent biconvex circular-arc sections was oscillated at an amplitude of 1.0 deg and a frequency of 21 cycles per second. The tests were made with the models at a mean angle of attack of 0 deg. The models were oscillated with a dynamic balance that was actuated by an electrohydraulic servo valve. The results of this investigation indicate the usefulness of this new apparatus. The experimental results of a previous damping-in-pitch investigation conducted in the Ames 6- by 6-foot supersonic wind tunnel at Mach numbers from 1.2 to 1.7 are included along with the theoretical results for this Mach number range. In the region of Mach numbers available for comparison, good agreement is shown to exist between the data obtained in the two facilities, except for some inconsistency in the slopes of the curves at $M = 1.2$ for the triangular wing. The results of this investigation clearly show that for the models tested the maximum values of the damping in pitch occur at Mach numbers very close to 1.0, and that abrupt changes in the pitch damping are encountered near sonic velocity.

Author

Wind Tunnel Tests; Body-Wing Configurations; Transonic Speed; Swept Wings; Dynamic Stability; Delta Wings; Rectangular Wings; Damping; Dimensional Stability

19980228316 NASA Langley Research Center, Hampton, VA USA

Influence of Large Positive Dihedral on Heat Transfer to Leading Edges of Highly Swept Wings at Very High Mach Numbers

Cooper, Morton, NASA Langley Research Center, USA; Stainback, P. Calvin, NASA Langley Research Center, USA; Apr. 1959; 20p; In English

Report No.(s): NASA-MEMO-3-7-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A geometric study has been made of some of the effects of dihedral on the heat transfer to swept delta wings. The results of this study show that the incorporation of large positive dihedral on highly swept wings can shift, even at moderately low angles of attack, the stagnation-line heat-transfer problem from the leading edges to the axis of symmetry (ridge line). An order-of-magnitude analysis (assuming laminar flow) indicates conditions for which it may be possible to reduce the heating at the ridge line (except in the vicinity of the wing apex) to a small fraction of the leading-edge heat transfer of a flat wing at the same lift. Furthermore, conditions are indicated where dihedral reduces the leading-edge heat transfer for angles of attack less than those required to shift the stagnation line from the leading edge to the ridge line.

Author

Aerodynamic Heat Transfer; Aerothermodynamics; Delta Wings; Dihedral Angle; Aerodynamic Heating; Leading Edges; Laminar Flow; Swept Wings; Aircraft Configurations

19980228317 NASA Ames Research Center, Moffett Field, CA USA

Full-Scale Wind-Tunnel Investigation of a Jet Flap in Conjunction with a Plain Flap with Blowing Boundary-Layer Control on a 35 deg Sweptback-Wing Airplane

Aoyagi, Kiyoshi, NASA Ames Research Center, USA; Hickey, David H., NASA Ames Research Center, USA; Mar. 1959; 28p; In English

Report No.(s): NASA-MEMO-2-20-59A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Previous investigations have shown that increased blowing at the hinge-line radius of a plain flap will give flap lift increases above that realized with boundary-layer control. Other experiments and theory have shown that blowing from a wing trailing edge,

through the jet flap effect, produced lift increases. The present investigation was made to determine whether blowing simultaneously at the hinge-line radius and trailing edge would be more effective than blowing separately at either location. The tests were made at a Reynolds number of 4.5×10^6 with a 35 deg sweptback-wing airplane. For this report, only the lift data are presented. Of the three flap blowing arrangements tested, blowing distributed between the trailing edge and the hinge-line radius of a plain flap was found to be superior to blowing at either location separately at the plain flap deflections of interest. Comparison of estimated and experimental jet flap effectiveness was fair.

Author

Boundary Layer Control; Sweptback Wings; Jet Flaps; Blowing; Externally Blown Flaps; Trailing Edge Flaps; Wind Tunnel Tests; Lift Augmentation

19980228319 NASA Langley Research Center, Hampton, VA USA

Static Lift, Drag, and Pitching-Moment Characteristics of Wings with Arrow and Modified-Diamond Plan Forms Combined with Several Different Bodies at Mach Numbers of 2.97, 3.35 and 3.71

Hasson, Dennis F., NASA Langley Research Center, USA; Presnell, John G., Jr., NASA Langley Research Center, USA; Jan. 1959; 52p; In English

Report No.(s): NASA-MEMO-1-24-59L; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Static lift, drag, and pitching-moment characteristics have been obtained at Mach numbers of 2.97, 3.35, and 3.71 for several wing-body combinations employing arrow- and modified-diamond-plan-form wings with about 70.3 deg of leading-edge sweep. The wings had uncambered, cambered, and cambered and twisted airfoil sections. The Reynolds numbers were approximately 2×10^6 and 3×10^6 for the wings with arrow- and modified-diamond plan forms, respectively. For arrow-wing configurations of the type tested, the uncambered wings led to greater maximum lift-drag ratios than did the wings with 20-percent leading-edge conical camber. All configurations tested were longitudinally stable throughout the test angle-of-attack and Mach number range. The static longitudinal stability parameter $(\Delta C_{m})/(\Delta C_L)$ was almost invariant with Mach number.

Author

Pitching Moments; Body-Wing Configurations; Angle of Attack; Static Stability; Low Aspect Ratio Wings; Lift Drag Ratio; Leading Edges; Swept Wings

19980228324 NASA Ames Research Center, Moffett Field, CA USA

The Effect of Leading-Edge Sweep and Surface Inclination on the Hypersonic Flow Field Over a Blunt Flat Plate

Creager, Marcus O., NASA Ames Research Center, USA; Jan. 1959; 50p; In English

Report No.(s): NASA-MEMO-12-26-58A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation of the effects of variation of leading-edge sweep and surface inclination on the flow over blunt flat plates was conducted at Mach numbers of 4 and 5.7 at free-stream Reynolds numbers per inch of 6,600 and 20,000, respectively. Surface pressures were measured on a flat plate blunted by a semicylindrical leading edge over a range of sweep angles from 0 deg to 60 deg and a range of surface inclinations from -10 deg to +10 deg. The surface pressures were predicted within an average error of ± 8 percent by a combination of blast-wave and boundary-layer theory extended herein to include effects of sweep and surface inclination. This combination applied equally well to similar data of other investigations. The local Reynolds number per inch was found to be lower than the free-stream Reynolds number per inch. The reduction in local Reynolds number was mitigated by increasing the sweep of the leading edge. Boundary-layer thickness and shock-wave shape were changed little by the sweep of the leading edge.

Author

Leading Edge Sweep; Supersonic Speed; Flat Plates; Flow Distribution; Slopes

19980228325 NASA Ames Research Center, Moffett Field, CA USA

Lift, Drag, and Pitching Moment of an Aspect-Ratio-2 Triangular Wing with Leading-Edge Flaps Designed to Simulate Conical Camber

Menees, Gene P., NASA Ames Research Center, USA; Dec. 1958; 32p; In English

Report No.(s): NASA-MEMO-10-5-58A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation was conducted to determine the effectiveness of leading-edge flaps in reducing the drag at lifting conditions of a triangular wing of aspect ratio 2.0. The flaps, deflected to simulate conically cambered wings having a wide range of design lift coefficients, were tested over a Mach number range of 0.70 to 2.22 through an angle-of-attack variation from -6 deg to +18 deg at a constant Reynolds number of 3.68 million based on the wing mean aerodynamic chord. A symmetrical wing of the same plan form and aspect ratio was also tested to provide a basis for comparison. The experimental results showed that with the flaps

in the undeflected position, a small amount of fixed leading-edge droop incorporated over the outboard 5 percent of the wing semi-span was as effective at high subsonic speeds as conical camber in improving the maximum lift-drag ratio above that of the symmetrical wing. At supersonic speeds, the penalty in minimum drag above that of the symmetrical wing was less than that incurred by conical camber. Deflecting the leading-edge flaps about the hinge line through 80 percent of the wing semispan resulted in further improvements of the drag characteristics at lift coefficients above 0.20 throughout the Mach number range investigated. The lift and pitching-moment characteristics were not significantly affected by the leading-edge flaps.

Author

Lift Drag Ratio; Pitching Moments; Aspect Ratio; Delta Wings; Leading Edge Flaps; Design Analysis; Simulation; Conical Camber; Experimentation

19980228346 NASA Langley Research Center, Hampton, VA USA

Aerodynamic Characteristics of a Model of a Proposed Six-Engine Hull-Type Seaplane at Mach Numbers of 1.57, 1.87, and 2.16

Morgan, James R., NASA Langley Research Center, USA; Fichter, Ann B., NASA Langley Research Center, USA; Apr. 1960; 43p; In English

Report No.(s): NASA-TM-X-271; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation was conducted in the Langley Unitary Plan wind tunnel to determine the aerodynamic performance and the longitudinal and lateral stability characteristics of a model of a proposed six-engine hull-type seaplane at Mach numbers of 1.57, 1.87, and 2.16. The results of the investigation indicated that the complete model configurations were longitudinally stable throughout the Mach number range. A marked change in static margin occurred at the higher Mach numbers, probably because of the intersection of the wing trailing-edge expansion wave with the horizontal stabilizer. The maximum lift-drag ratio was 4.4 at Mach numbers of 1.57 and 1.87 and decreased to a value of 4.2 at a Mach number of 2.16. A stable variation in the lateral and directional stability derivatives was evident throughout the Mach number range.

Author

Aerodynamic Characteristics; Longitudinal Stability; Lateral Stability; Supersonic Speed; Wind Tunnel Tests; Seaplanes; Aircraft Design; Aircraft Engines

19980228351 NASA Ames Research Center, Moffett Field, CA USA

Experimental and Theoretical Study of a Rectangular Wing in a Vortical Wake at Low Speed

Smith, Willard G., NASA Ames Research Center, USA; Lazzeroni, Frank A., NASA Ames Research Center, USA; Oct. 1960; 38p; In English

Report No.(s): NASA-TN-D-339; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A systematic study has been made, experimentally and theoretically, of the effects of a vortical wake on the aerodynamic characteristics of a rectangular wing at subsonic speed. The vortex generator and wing were mounted on a reflection plane to avoid body-wing interference. Vortex position, relative to the wing, was varied both in the spanwise direction and normal to the wing. Angle of attack of the wing was varied from -40 to +60. Both chordwise and spanwise pressure distributions were obtained with the wing in uniform and vortical flow fields. Stream surveys were made to determine the flow characteristics in the vortical wake. The vortex-induced lift was calculated by several theoretical methods including strip theory, reverse-flow theory, and reverse-flow theory including a finite vortex core. In addition, the Prandtl lifting-line theory and the Weissinger theory were used to calculate the spanwise distribution of vortex-induced loads. With reverse-flow theory, predictions of the interference lift were generally good, and with Weissinger's theory the agreement between the theoretical spanwise variation of induced load and the experimental variation was good. Results of the stream survey show that the vortex generated by a lifting surface of rectangular plan form tends to trail back streamwise from the tip and does not approach the theoretical location, or centroid of circulation, given by theory. This discrepancy introduced errors in the prediction of vortex interference, especially when the vortex core passed immediately outboard of the wing tip. The wake produced by the vortex generator in these tests was not fully rolled up into a circular vortex, and so lacked symmetry in the vertical direction of the transverse plane. It was found that the direction of circulation affected the induced loads on the wing either when the wing was at angle of attack or when the vortex was some distance away from the plane of the wing.

Author

Vortex Generators; Rectangular Wings; Aerodynamic Characteristics; Body-Wing Configurations; Flow Theory

19980228353 NASA Ames Research Center, Moffett Field, CA USA

The Stabilizing Effectiveness of Conical Flares on Bodies with Conical Noses

Kirk, Donn B., NASA Ames Research Center, USA; Chapman, Gary T., NASA Ames Research Center, USA; Sep. 1959; 26p; In English

Report No.(s): NASA-TM-X-30; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An analysis is presented of published results of force tests on 80 cone-cylinder-flare configurations at Mach numbers of 2.18, 2.81, and 4.04. The contributions, excluding interference effects, of the cone-cylinder bodies to the over-all normal force derivatives have been removed by means of the second-order shock-expansion method, and the normal force derivatives at zero angle of attack due to the flares alone are shown. The results from a wide variety of configurations are correlated by plotting ratios of the normal force derivatives of the flares to the normal force derivatives of cones having the same included angle. Comparisons are made of the experimental normal force results with the normal force derivatives obtained by assuming conical flow over the flares and with those obtained by use of the second-order shock-expansion method. The comparisons show that use of the second-order shock-expansion method is generally the superior of the two, and in most cases gives values of the normal force derivatives of the flares which agree very well with the experimental results. Centers of pressure of the flares are presented and comparisons are made with results obtained from the theories mentioned. In general, the comparisons show that the assumption of conical flow over the flares is comparable to use of the second-order shock-expansion method in determining the centers of pressure, and in many cases both methods give values which agree closely with the experimental results.

Author

Aerodynamic Configurations; Conical Flow; Zero Angle of Attack; Conical Bodies

19980228365 NASA Langley Research Center, Hampton, VA USA

Low-Speed Wind-Tunnel Investigation to Determine the Aerodynamic Characteristics of a Rectangular Wing Equipped with a Full-Span and an Inboard Half-Span Jet-Augmented Flap Deflected 55 deg

Gainer, Thomas G., NASA Langley Research Center, USA; Feb. 1959; 24p; In English

Report No.(s): NASA-MEMO-1-27-59L; L-156; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation to determine the aerodynamic characteristics of a rectangular wing equipped with a full-span and an inboard half-span jet-augmented flap has been made in the Langley 300 MPH 7- by 10-foot tunnel. The wing had an aspect ratio of 8.3 and a thickness-chord ratio of 0.167. A jet of air was blown backward through a small gap, tangentially to the upper surface of a round trailing edge, and was separated from the trailing edge by a very small flap at an angle of 55 deg with respect to the wing-chord plane. The results of the investigation showed that the ratio of total lift to jet-reaction lift for the wing was about 35 percent less for the half-span jet-augmented flap than for the full-span jet-augmented flap. The reduction of the span of the jet-augmented flap from full to half span reduced the maximum value of jet-circulation lift coefficient that could be produced from about 6.8 to a value of about 2.2. The half-span jet-augmented flap gave thrust recoveries considerably poorer than those obtained with the full-span jet-augmented flap. Large nose-down pitching-moment coefficients were produced by the half-span flap, with the greater part of these being the result of the larger jet reactions required to produce a given lift for the half-span flap compared with that required for the full-span flap.

Author

Wind Tunnel Tests; Rectangular Wings; Flapping; Jet Flaps; Jet Lift; Lift Devices; Aerodynamic Characteristics; Air Jets

19980228366 NASA Langley Research Center, Hampton, VA USA

Free-Spinning-Tunnel Investigation of a 1/40-Scale Model of the McConnell F-101A Airplane

Bowman, James S., Jr., NASA Langley Research Center, USA; Healy, Frederick M., NASA Langley Research Center, USA; 1959; 14p; In English

Report No.(s): NASA-MEMO-3-14-59L; AF-AM-87; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been made in the Langley 20-foot free-spinning tunnel of a 1/40-scale model of the McDonnell F-101A airplane to alleviate the unfavorable spinning characteristics encountered with the airplane. The model results indicate that a suitable strake extended on the inboard side of the nose of the airplane (right side in a right spin) in conjunction with the use of optimum control recovery technique will terminate spin rotation of the airplane. It may be difficult to recover from subsequent high angle-of-attack trimmed flight attitudes even by forward stick movement. The optimum spin-recovery control technique for the McDonnell F-101A is simultaneous full rudder reversal to against the spin and aileron movement to full with the spin (stick full right in a right erect spin) and forward movement of the stick immediately after rotation stops.

Author

Aircraft Spin; Scale Models; Aerodynamic Characteristics; Optimal Control; Rotation; Strakes; Controllability; Spin Dynamics; Wind Tunnel Tests

19980228368 NASA Ames Research Center, Moffett Field, CA USA

Low-Speed Tests of Semispan-Wing Models at Angles of Attack from 0 to 180 degrees

Koenig, David G., NASA Ames Research Center, USA; Apr. 1959; 26p; In English

Report No.(s): NASA-MEMO-2-27-59A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Semispan-wing models were tested at angles of attack from 0 to 180 deg at low subsonic speeds. Eight plan forms were considered, both swept and unswept with aspect ratios ranging from 2 to 6. Except for a delta-wing model of aspect ratio 2, all models had a taper ratio of 0.5 and an NACA 64A010 airfoil section. The delta-wing model had an NACA 0005 (modified) airfoil section. With two exceptions, the models were tested both with and without a full-span trailing-edge flap deflected 25 deg. The Reynolds numbers based on the mean aerodynamic chord were between 1.5 and 2.2 million. Lift, drag, and pitching-moment coefficients are presented as functions of angle of attack. Approximate corrections for the effects of blockage were applied to the data.

Author

Aerodynamic Coefficients; Wind Tunnel Tests; Airfoil Profiles; Angle of Attack; Delta Wings; Semispan Models; Subsonic Speed

19980228393 NASA Langley Research Center, Hampton, VA USA

Full-Scale Wind-Tunnel Investigation of the VZ-5 Four-Propeller Deflected-Slipstream VTOL Airplane

Fink, Marvin P., NASA Langley Research Center, USA; Feb. 20, 1963; 32p; In English

Report No.(s): NASA-TM-SX-805; L-3059; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The investigation was conducted to determine the static stability and control characteristics of the VZ-5 VTOL airplane over the speed range from hovering to forward flight. Force and moment data were taken over a range of angles of attack of 0 to 15 deg and a range of sideslip of +/-10 deg for flap deflections from 0 to 77 deg. The longitudinal stability and trim characteristics were found to be quite unacceptable and it did not seem that they could be corrected with any reasonable modifications to the airplane.

Author

Static Stability; Propeller Slipstreams; Flapping; Longitudinal Stability; Vertical Takeoff Aircraft; Wind Tunnel Tests; Sideslip; Scale Models; Directional Stability

19980228400 NASA Langley Research Center, Hampton, VA USA

Charts of the Induced Velocities Near a Lifting Rotor

Jewel, Joseph W., Jr., NASA Langley Research Center, USA; Heyson, Harry H., NASA Langley Research Center, USA; May 1959; 68p; In English

Report No.(s): NASA-MEMO-4-15-59L; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

A compilation of charts of the induced velocities near a lifting rotor is presented. The charts cover uniform as well as various non-uniform distributions of disk loading and should be applicable to many aerodynamic interference problems involving rotors.

Author

Velocity; Lifting Rotors; Interactional Aerodynamics; Rotor Aerodynamics

19980228403 NASA Langley Research Center, Hampton, VA USA

Free-Spinning-Tunnel Investigation of a 1/17 Scale Model of the Cessna T-37A Airplane

Bowman, James S., Jr., NASA Langley Research Center, USA; Healy, Frederick M., NASA Langley Research Center, USA; 1958; 16p; In English

Report No.(s): NASA-MEMO-3-1-59L; AF-AM-42; L-237; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Results of an investigation of a dynamic model in the Langley 20-foot free-spinning tunnel are presented. Erect spin and recovery characteristics were determined for a range of mass distributions and center-of-gravity positions. The effects of lateral displacement of the center of gravity, engine rotation, nose strakes, and increased rudder area were investigated.

Author

Wind Tunnel Tests; Spin Tests; Cessna Aircraft; Scale Models

19980228433 NASA Ames Research Center, Moffett Field, CA USA

A Technique for Determining Relaxation Times by Free-Flight Tests of Low-Fineness-Ratio Cones; With Experimental Results for Air at Equilibrium Temperatures up to 3440 K

Stephenson, Jack D., NASA Ames Research Center, USA; Sep. 1960; 58p; In English

Report No.(s): NASA-TN-D-327; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

This report describes a technique which combines theory and experiments for determining relaxation times in gases. The technique is based on the measurement of shapes of the bow shock waves of low-fineness-ratio cones fired from high-velocity guns.

The theory presented in the report provides a means by which shadowgraph data showing the bow waves can be analyzed so as to furnish effective relaxation times. Relaxation times in air were obtained by this technique and the results have been compared with values estimated from shock tube measurements in pure oxygen and nitrogen. The tests were made at velocities ranging from 4600 to 12,000 feet per second corresponding to equilibrium temperatures from 35900 R (19900 K) to 6200 R (34400 K), under which conditions, at all but the highest temperatures, the effective relaxation times were determined primarily by the relaxation time for oxygen and nitrogen vibrations.

Author

Bow Waves; Shock Waves; Shadowgraph Photography; Estimating; Fineness Ratio; Nitrogen; Oxygen

19980228464 NASA Langley Research Center, Hampton, VA USA

Low Subsonic Pressure Distributions on Three Rigid Wings Simulating Paragliders with Varied Canopy Curvature and Leading-Edge Sweep

Fournier, Paul G., NASA Langley Research Center, USA; Bell, B. Ann, NASA Langley Research Center, USA; Nov. 1961; 54p; In English

Report No.(s): NASA-TN-D-983; L-1757; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An investigation has been made in the Langley 7- by 10-foot transonic tunnel to determine the subsonic pressure distribution of three paraglider models through an angle-of-attack range from 0 deg to 74 deg. Three rigid metal models simulated a 45 deg basic flat planform paraglider with leading-edge sweep angles of 61.6 deg, 52.5 deg, and 48.6 deg. These configurations resulted in one-half-circle, one-third-circle, and one-quarter-circle semispan trailing-edge curvature when viewed from downstream. The results of the investigation are presented as curves of chordwise pressure distributions at four spanwise locations.

Author

Leading Edge Sweep; Circles (Geometry); Paragliders; Pressure Distribution; Planforms; Rigid Wings

19980228466 NASA Langley Research Center, Hampton, VA USA

Effect of Afterbody-Ejector Configurations on the Performance at Transonic Speeds of a Pylon-Supported Nacelle Model having a Hot-Jet Exhaust

Swihart, John M., NASA Langley Research Center, USA; Mercer, Charles E., NASA Langley Research Center, USA; Norton, Harry T., Jr., NASA Langley Research Center, USA; Feb. 1959; 56p; In English

Report No.(s): NASA-MEMO-1-4-59L; L-133; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An investigation of several afterbody-ejector configurations on a pylon-supported nacelle model has been completed in the Langley 16-foot transonic tunnel at Mach numbers from 0.80 to 1.05. The propulsive performance of two nacelle afterbodies with low boattailing and long ejector spacing was compared with a configuration corresponding to a turbojet-engine installation having a highly boattailed afterbody with a short ejector. The jet exhaust was simulated with a hydrogen peroxide turbojet simulator. The angle of attack was maintained at 0 deg, and the average Reynolds number based on body length was 20×10^6 . The results of the investigation indicated that the configuration with a conical afterbody with smooth transition to a 15 deg boattail angle had large beneficial jet effects on afterbody pressure-drag coefficient and had the best thrust-minus-drag performance of the afterbody-ejector configurations investigated.

Author

Nacelles; Aerodynamic Configurations; Aerodynamic Drag; High Temperature Gases; Boattails; Ejectors; Afterbodies; Jet Exhaust

19980228469 NASA Lewis Research Center, Cleveland, OH USA

Effect on Inlet Performance of a Cowl Visor and an Internal-Contraction Cowl for Drag Reduction at Mach Numbers 3.07 and 1.89

Gertsma, Laurence W., NASA Lewis Research Center, USA; Apr. 1959; 26p; In English

Report No.(s): NASA-MEMO-3-18-59E; E-173; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Two methods for reducing the external cowl angle, and hence the cowl pressure drag, were investigated on a two-dimensional model. One method used at both on- and off-design Mach numbers was the addition of a cowl visor that had the inner surface parallel to the free stream at 0 deg angle of attack. The other method investigated consisted in replacing the original cowl by a flatter cowl that also provided internal contraction. Both the visor and the internal-contraction cowl reduced the cowl pressure drag 64 percent or more. The visor had little effect on inlet performance at the design Mach number except to reduce the stability range slightly. At off-design, the visor caused an increase in critical pressure recovery.

Author

Two Dimensional Models; Drag Reduction; Pressure Drag; Stability; Critical Pressure; Engine Inlets

19980228471 NASA Langley Research Center, Hampton, VA USA

Wind-Tunnel Investigation at Mach Numbers from 0.40 to 1.14 of the Static Aerodynamic Characteristics of a Nonlifting Vehicle Suitable for Reentry

Pearson, Albin O., NASA Langley Research Center, USA; May 1959; 14p; In English

Report No.(s): NASA-MEMO-4-13-59L; L-437; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation was conducted in the Langley 8-foot transonic pressure tunnel to investigate the static pitching-moment, normal-force, and axial-force characteristics on a model of a nonlifting vehicle suitable for reentry. The vehicle was designed to use a heat sink and blunt shape to alleviate the effects of the heating encountered during reentry of the earth's atmosphere. The effects of modifying the intersection of the face of the model with the afterbody from a sharp corner to a rounded edge were also investigated. Tests were conducted at Mach numbers from 0.40 to 1.14 and at angles of attack from approximately -3 deg to 20 deg. The Reynolds number varied from about 2.0×10^6 to 3.6×10^6 . The results show that the model had a low positive static-stability level, low normal-force coefficients, and large axial-force coefficients. The model trimmed, for the angle-of-attack range investigated, at angles of attack near zero. The effects on the stability as a result of rounding the corner were negligible.

Author

Wind Tunnel Tests; Mach Number; Static Aerodynamic Characteristics; Research; Experimentation; Pitching Moments; Angle of Attack

19980228476 NASA Ames Research Center, Moffett Field, CA USA

An Investigation of the Drag Characteristics of a Tailless Delta-Wing Airplane in Flight, Including Comparison with Wind-Tunnel Data

Rolls, L. Stewart, NASA Ames Research Center, USA; Wingrove, Rodney C., NASA Ames Research Center, USA; Nov. 1958; 36p; In English

Report No.(s): NASA-MEMO-10-8-58A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A series of flight tests were conducted to determine the lift and drag characteristics of an F4D-1 airplane over a Mach number range of 0.80 to 1.10 at an altitude of 40,000 feet. Apparently satisfactory agreement was obtained between the flight data and results from wind-tunnel tests of an 0.055-scale model of the airplane. Further tests show the apparent agreement was a consequence of the altitude at which the first tests were made.

Author

Drag; Experimentation; Delta Wings; Aircraft Models

19980228494 Dayton Univ. Research Inst., Structural Integrity Div., OH USA

Statistical Loads Data for Boeing 737-400 Aircraft in Commercial Operations Final Report

Rustenburg, John, Dayton Univ. Research Inst., USA; Skinn, Donal, Dayton Univ. Research Inst., USA; Tipps, Daniel O., Dayton Univ. Research Inst., USA; Aug. 1998; 96p; In English

Contract(s)/Grant(s): FAA-96-G-020

Report No.(s): PB98-174543; UDR-TR-98-00032; DOT/FAA/AR-98/28; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

The primary objective of this research is to support the FAA Airborne Data Monitoring Systems Research Program by developing new and improved methods and criteria for processing and presenting large commercial transport airplane flight and ground loads usage data. The scope of activities performed involved (1) defining the service related factors which affect the operational life of commercial aircraft; (2) designing an efficient software system to reduce, store, and process large quantities of optical quick access recorder data; and (3) providing processed data in formats that will enable the FAA to reassess existing certification criteria. Presented are analyses and statistical summaries of data collected from 11,721 flights representing 19,105 flight hours of 17 typical B-737-400 aircraft during operational usage recorded by a single airline. The data include statistical information on accelerations, speeds, altitudes, flight duration and distance, gross weights, speed brake/spoiler cycles, thrust reverser usage, and gust velocities encountered.

NTIS

Loads (Forces); Data Systems; Data Acquisition; Boeing 737 Aircraft; Flight Load Recorders

19980230611 NASA Langley Research Center, Hampton, VA USA

Effect of Nose Length, Fuselage Length, and Nose Fineness Ratio on the Longitudinal Aerodynamic Characteristics of Two Complete Models at High Subsonic Speeds

Goodson, Kenneth W., NASA Langley Research Center, USA; Oct. 1958; 44p; In English

Report No.(s): NASA-MEMO-10-10-58L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been made of the effects of nose length, fuselage length, and nose fineness ratio on the static longitudinal aerodynamic characteristics of an airplane model with a swept wing and low tail and of a second model with a highly tapered wing of moderate sweep and a T-tail. The tests were conducted in the Langley high-speed 7- by 10-foot tunnel at Mach numbers from 0.60 to 0.92. The nose and body cross sections were circular. For either the model with the swept wing and low tail or the model with the highly tapered wing of moderate sweep and the T-tail, the effects of forebody changes amounted primarily to rotations of the pitching-moment curves (changes in static margin) over the test ranges of angle of attack and Mach number. For the range of body shapes investigated the longitudinal stability at low lift is decreased by an increase in nose length or in fuselage length or by a reduction in nose fineness ratio when the fuselage length is held constant. In general, the stability for all model configurations showed substantially the same variation with changes in forebody area moment. The forebody changes did not alter the angle of attack at which an unstable break occurred in the moment contribution of the T-tail but did alter somewhat the magnitude of the instability.

Author

Noise Intensity; Fuselages; Longitudinal Stability; Subsonic Speed; Static Aerodynamic Characteristics; Pitching Moments; Aircraft Models

19980230617 NASA, Washington, DC USA

On the Relation Between the Generation of a Lift Force on a Wing and the Character of the Flow in the Boundary Layer

Ostoslavskii, I. V.; Grumondz, T. A.; *Izvestiia vysshikh uchebnykh zavedenii, Seriya Aviatsionnaia Tekhnika*; May 1960, No. 1, pp. 27-36; In English

Report No.(s): NASA-TT-F-26; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

The paper presents the results of a theoretical and experimental investigation of the mechanism involved in the generation of lift on a wing as the wing is started from rest. It is shown theoretically that as the wing motion as started, vortices form near the wing leading edge on both surfaces due to viscous effect, and these vortices from flow downstream. At angle of attack the vortices on the lower surface reach the trailing edge first, leaving a surplus of vortices on the upper surface after steady motion is reached which are related to the general circulation or lift. A comparison of experiment with theory is included.

Author

Leading Edges; Numerical Analysis; Data Acquisition; Experimentation; Vortices; Lift

19980230620 NASA Langley Research Center, Hampton, VA USA

Transonic and Supersonic Wind-Tunnel Tests of Wing-Body Combinations Designed for High Efficiency at a Mach Number of 1.41

Grant, Frederick C., NASA Langley Research Center, USA; Sevier, John R., Jr., NASA Langley Research Center, USA; Oct. 1960; 84p; In English

Report No.(s): NASA-TN-D-435; L-260; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

Wind-tunnel force tests of a number of wing-body combinations designed for high lift-drag ratio at a Mach number of 1.41 are reported. Five wings and six bodies were used in making up the various wing-body combinations investigated. All the wings had the same highly swept continuously tapered plan form with NACA 65A-series airfoil sections 4 percent thick at the root tapering linearly to 3 percent thick at the tip. The bodies were based on the area distribution of a Sears-Haack body of revolution for minimum drag with a given length and volume. These wings and bodies were used to determine the effects of wing twist, wing twist and camber, wing leading-edge droop, a change from circular to elliptical body cross-sectional shape, and body indentation by the area-rule and streamline methods. The supersonic test Mach numbers were 1.41 and 2.01. The transonic test Mach number range was from 0.6 to 1.2. For the transition-fixed condition and at a Reynolds number of 2.7×10^6 based on the mean aerodynamic chord, the maximum value of lift-drag ratio at a Mach number of 1.41 was 9.6 for a combination with a twisted wing and an indented body of elliptical cross section. The tests indicated that the transonic rise in minimum drag was low and did not change appreciably up to the highest test Mach number of 2.01. The lower values of lift-drag ratio obtained at a Mach number of 2.01 can be attributed to the increase of drag due to lift with Mach number.

Author

Wind Tunnel Tests; Supersonic Speed; Transonic Speed; Reynolds Number; Lift Drag Ratio; Airfoil Profiles; Body-Wing Configurations

19980230621 NASA Langley Research Center, Hampton, VA USA

Hovering and Transition Flight Tests of a 1/5-Scale Model of a Jet-Powered Vertical-Attitude VTOL Research Airplane

Smith, Charles C., Jr., NASA Langley Research Center, USA; May 1961; 32p; In English

Report No.(s): NASA-TN-D-404; L-1555; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An experimental investigation has been made to determine the dynamic stability and control characteristics of a 1/5-scale flying model of a jet-powered vertical-attitude VTOL research airplane in hovering and transition flight. The model was powered with either a hydrogen peroxide rocket motor or a compressed-air jet exhausting through an ejector tube to simulate the turbojet engine of the airplane. The gyroscopic effects of the engine were simulated by a flywheel driven by compressed-air jets. In hovering flight the model was controlled by jet-reaction controls which consisted of a swiveling nozzle on the main jet and a movable nozzle on each wing tip; and in forward flight the model was controlled by elevons and a rudder. If the gyroscopic effects of the jet engine were not represented, the model could be flown satisfactorily in hovering flight without any automatic stabilization devices. When the gyroscopic effects of the jet engine were represented, however, the model could not be controlled without the aid of artificial stabilizing devices because of the gyroscopic coupling of the yawing and pitching motions. The use of pitch and yaw dampers made these motions completely stable and the model could then be controlled very easily. In the transition flight tests, which were performed only with the automatic pitch and yaw dampers operating, it was found that the transition was very easy to perform either with or without the engine gyroscopic effects simulated, although the model had a tendency to fly in a rolled and sideslipped attitude at angles of attack between approximately 25 deg and 45 deg because of static directional instability in this range.

Author

Hovering; Angle of Attack; Vertical Takeoff Aircraft; Horizontal Flight; Flight Tests; Directional Stability; Dynamic Stability

19980230625 NASA, Washington, DC USA

Effect of Slight Blunting of Leading Edge of an Immersed Body on the Flow Around it at Hypersonic Speeds

Chernyi, G. G.; Izvestiia Akademia Nauk, Otdelenie Tekhnicheskikh Nauk; Jun. 1960, No. 4, pp. 54-66; In English; Translated by Consultants Custom Translations, Inc., 227 West 17th Street, New York, NY

Report No.(s): NASA-TT-F-35; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A theoretical investigation of the flow about blunt leading-edge bodies, immersed in a hypersonic stream of any gas is undertaken. The study utilizes piston and blast-wave theory to study the flow about such shapes as a blunt leading-edge slab, a blunt-nosed cylindrical rod, a thin wedge with a blunt leading edge, and a thin blunt-nosed cone. Expressions are obtained for predicting the surface pressure distribution, the shape of the bow shock, and the drag of these bodies, and these theoretical results are compared with experimental results for several of the cases considered.

Author

Blunt Leading Edges; Hypersonic Flow; Hypersonic Speed; Leading Edges; Pressure Distribution; Shock Waves; Detonation Waves

19980230630 NASA Lewis Research Center, Cleveland, OH USA

Performance Characteristics of Flush and Shielded Auxiliary Exits at Mach Numbers of 1.5 to 2.0

Abdalla, Kaleel L., NASA Lewis Research Center, USA; Jun. 1959; 24p; In English

Report No.(s): NASA-MEMO-5-18-59E; E-139; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The performance characteristics of several flush and shielded auxiliary exits were investigated at Mach numbers of 1.5 to 2.0, and jet pressure ratios from jet off to 10. The results indicate that the shielded configurations produced better overall performance than the corresponding flush exits over the Mach-number and pressure-ratio ranges investigated. Furthermore, the full-length shielded exit was highest in performance of all the configurations. The flat-exit nozzle block provided considerably improved performance compared with the curved-exit nozzle block.

Author

Performance Tests; Auxiliary Propulsion; Data Acquisition

19980230644 NASA Langley Research Center, Hampton, VA USA

Supersonic Free-Flight Measurements of Heat Transfer and Transition on a 10 degree Cone having a Low Temperature Ratio

Merlet, Charles F., NASA Langley Research Center, USA; Rumsey, Charles B., NASA Langley Research Center, USA; Aug. 1961; 28p; In English

Report No.(s): NASA-TN-D-951; L-1700; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Heat-transfer coefficients in the form of Stanton number and boundary-layer transition data were obtained from a free-flight test of a 100-inch-long 10 deg. total-angle cone with a 1/16-inch tip radius which penetrated deep into the region of infinite stability of laminar boundary layer over a range of wall-to-local-stream temperature ratio and for local Mach numbers from 1.8 to 3.5. Experimental heat-transfer coefficients, obtained at Reynolds numbers up to 160×10^6 , were in general somewhat higher than theoretical values. A maximum Reynolds number of transition of only 33×10^6 was obtained. Contrary to theoretical

and some other experimental investigations, the transition Reynolds number initially increased while the wall temperature ratio increased at relatively constant Mach number. Further increases in wall temperature ratio were accompanied by a decrease in transition Reynolds number. Increasing transition Reynolds number with increasing Mach number was also indicated at a relatively constant wall temperature ratio.

Author

Aerothermodynamics; Boundary Layer Transition; Supersonic Heat Transfer; Stanton Number; Wind Tunnel Tests; Reynolds Number; Heat Transfer Coefficients; Circular Cones

19980230676 NASA Langley Research Center, Hampton, VA USA

Aerodynamic Effects of Some Configuration Variables on the Aeroelastic Characteristics of Lifting Surfaces at Mach Numbers from 0.7 to 6.86

Hanson, Perry W., NASA Langley Research Center, USA; Nov. 1961; 52p; In English

Report No.(s): NASA-TN-D-984; L-1626; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Results of flutter tests on some simple all-movable-control-type models are given. One set of models, which had a square planform with double-wedge airfoils with four different values of leading- and trailing-edge radii from 0 to 6 percent chord and airfoil thicknesses of 9, 11, 14, and 20 percent chord, was tested at Mach numbers from 0.7 to 6.86. The bending-to-torsion frequency ratio was about 0.33. The other set of models, which had a tapered planform with single-wedge and double-wedge airfoils with thicknesses of 3, 6, 9, and 12 percent chord, was tested at Mach numbers from 0.7 to 3.98 and a frequency ratio of about 0.42. The tests indicate that, in general, increasing thickness has a destabilizing effect at the higher Mach numbers but is stabilizing at subsonic and transonic Mach numbers. Double-wedge airfoils are more prone to flutter than single-wedge airfoils at comparable stiffness levels. Increasing airfoil bluntness has a stabilizing effect on the flutter boundary at supersonic speeds but has a negligible effect at subsonic speeds. However, increasing bluntness may also lead to divergence at supersonic speeds. Results of calculations using second-order piston-theory aerodynamics in conjunction with a coupled-mode analysis and an uncoupled-mode analysis are compared with the experimental results for the sharp-edge airfoils at supersonic speeds. The uncoupled-mode analysis more accurately predicted the flutter characteristics of the tapered-planform models, whereas the coupled-mode analysis was somewhat better for the square-planform models. For both the uncoupled- and coupled-mode analyses, agreement with the experimental results improved with increasing Mach number. In general, both methods of analysis gave unconservative results with respect to the experimental flutter boundaries.

Author

Aerodynamics; Aeroelasticity; Flutter Analysis; Coupled Modes; Boundaries

19980230683 NASA Langley Research Center, Hampton, VA USA

Basic Pressure Measurements at Transonic Speeds on a Thin 45 deg Sweptback highly Tapered Wing With Systematic Spanwise Twist Variations

Mugler, John P., Jr., NASA Langley Research Center, USA; Apr. 1959; 150p; In English

Report No.(s): NASA-MEMO-2-24-59L; L-207; No Copyright; Avail: CASI; A07, Hardcopy; A02, Microfiche

Pressure distributions obtained in the Langley 8-foot transonic pressure tunnel on a thin highly tapered twisted 45 deg swept-back wing-body combination are presented. The wing has a quadratic spanwise twist variation from 0 deg at 10 percent of the semispan to 6 deg at the tip. The tip is at a lower angle of attack than the root. Tests were made at stagnation pressures of both 0.5 and 1.0 atmosphere at Mach numbers from 0.800 to 1.200 through an angle-of-attack range from -4 deg to 20 deg.

Author

Pressure Distribution; Angle of Attack; Body-Wing Configurations; Stagnation Pressure; Sweptback Wings

19980230685 NASA Langley Research Center, Hampton, VA USA

Investigation on the use of a Freely Rotating Rotor at the Cowl Face of a Supersonic Conical Inlet to Reduce Inlet Flow Distortion

Goldberg, Theodore J., NASA Langley Research Center, USA; Boxer, Emanuel, NASA Langley Research Center, USA; Jun. 1959; 48p; In English

Report No.(s): NASA-MEMO-5-28-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been made on the use of a freely rotating rotor at the cowl face of a supersonic conical diffuser to determine its effectiveness in reducing inlet flow distortion and the penalty in terms of total-pressure loss imposed by such a device when distortions are negligible. Tests were made with a rotor having an inlet tip diameter of 2.18 inches and a ratio of hub radius to tip radius of 0.52, in conjunction with a conical inlet having a 25 deg semi-vertex cone angle, at a Mach number of 2.1 over an angle-of-attack range of 0 deg to 8 deg. A simplified analysis showing that a supersonic, freely rotating rotor with maximum

solidity for noninterference between blades will operate in an undistorted flow with a total-pressure defect of 1 percent or less was experimentally verified. Overall total-pressure distortions of 0.1 to 0.4 and Mach number distortions of 0.4 to 1.4, obtained at 4 deg to 8 deg angle of attack, were reduced about 30 percent and 23 percent, respectively, because of the presence of the rotor, with no measurable total-pressure loss. The rotor increased the peak total-pressure recovery at the simulated combustion chamber 1 1/2 and 3 1/2 percent at 6 deg and 8 deg angles of attack, respectively. This increase is attributed to lower diffusion duct losses as a consequence of a more uniform flow created by the rotor.

Author

Research; Rotors; Rotating Bodies; Inlet Flow; Inlet Nozzles; Effectiveness; Flow Distortion

19980231027 NASA, Washington, DC USA

On the Computation of the Circulation of a Gliding Wing of Large Span

Monakhov, N. M.; Izvestiia Vysshikh Uchebnykh Zavedenii, Seriya, Aviatsionnaya Tekhnika; Jun. 1960; Volume 1, pp. 19-26; In English; Translated by Consultants Custom Translations, Inc., 227 West 17th Street, New York 11, NY

Report No.(s): NASA-TT-F-31; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A method is presented for the theoretical determination of the circulation of a large span swept wing using lifting surface theory. By solution of an integral equation, expressions are derived for the intensity of the wing vortex layer along and at right angles to the flow. The results are compared with those of the simpler but nonrigorous "three-quarter chord" method of determining the circulation. Use of the latter method is shown to be justified for swept wings of large span.

Author

Integral Equations; Swept Wings; Wing Span; Lift

19980231058 NASA Langley Research Center, Hampton, VA USA

Aerodynamic Characteristics of Low-Aspect-Ratio Wings in Close Proximity to the Ground

Fink, Marvin P., NASA Langley Research Center, USA; Lastinger, James L., NASA Langley Research Center, USA; Jul. 1961; 40p; In English

Report No.(s): NASA-TN-D-926; L-1367; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A wind-tunnel investigation has been conducted to determine the effect of ground proximity on the aerodynamic characteristics of thick highly cambered rectangular wings with aspect ratios of 1, 2, 4, and 6. The results showed that, for these aspect ratios, as the ground was approached all wings experienced increases in lift-curve slope and reductions in induced drag which resulted in increases in lift-drag ratio. Although an increase in lift-curve slope was obtained for all aspect ratios as the ground was approached, the lift coefficient at an angle of attack of 0 deg for any given aspect ratio remained nearly constant. The experimental results were in general agreement with Wieselsberger's ground-effect theory (NACA Technical Memorandum 77). As the wings approached the ground, there was an increase in static longitudinal stability at positive angles of attack. When operating in ground effect, all the wings had stability of height at positive angles of attack and instability of height at negative angles of attack. Wing-tip fairings on the wings with aspect ratios of 1 and 2 produced small increases in lift-drag ratio in ground effect. End plates extending only below the chord plane on the wing with an aspect ratio of 1 provided increases in lift coefficient and in lift-drag ratio in ground effect.

Author

Aerodynamic Characteristics; Low Aspect Ratio Wings; Wind Tunnel Tests; Aspect Ratio; Cambered Wings; Ground Effect (Aerodynamics); Rectangular Wings

19980231060 NASA Ames Research Center, Moffett Field, CA USA

An Investigation of the Pressure Distribution on a 1/15-Scale Model of the Lockheed WS-117L Vehicle Plus Booster "B" at Mach Numbers from 1.55 to 2.35

Martin, Norman J., NASA Ames Research Center, USA; Mar. 1959; 42p; In English

Report No.(s): NASA-MEMO-3-13-59A; AF-AM-163; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Pressure coefficients were measured over the vehicle and over the forward part of the booster at Reynolds numbers of 3.0×10^6 per foot. Tabular results are presented for two nose shapes at Mach numbers of 1.55, 1.75, 2.00, and 2.35, at angles of attack from -4 deg to +10 deg, and at 0 deg sideslip.

Author

Research; Experimentation; Pressure Distribution; Scale Models; Wind Tunnel Tests

19980231065 NASA Ames Research Center, Moffett Field, CA USA

An Experimental Investigation of the Pressure Distribution on A 1/15-Scale Model of the Lockheed WS-117L Vehicle Plus Booster "B" at Mach Numbers from 0.70 to 1.45

Fahey, Russell E., NASA Ames Research Center, USA; Marker, Ralph D., NASA Ames Research Center, USA; Mar. 1959; 67p; In English

Report No.(s): NASA-MEMO-3-12-59A; A-217; AF-AM-163; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Results obtained with two nose shapes tested at a Reynolds number per foot of 5×10^6 at angles of attack from -4 deg to +10 deg at 0 deg angle of sideslip are presented in tabulated pressure coefficient form without analysis.

Author

Experimentation; Pressure Distribution; Scale Models

19980231067 NASA Langley Research Center, Hampton, VA USA

Jet Interference Effects on a Model of a Single-Engine Four Jet V/STOL Airplane at Mach Numbers from 0.60 to 1.00

Schmeer, James W., NASA Langley Research Center, USA; Runckel, Jack F., NASA Langley Research Center, USA; 1962; 32p; In English

Report No.(s): NASA-TM-SX-685; L-2043; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation was conducted in the Langley 16-foot transonic tunnel to determine the interference from four exhaust jets on the aerodynamic characteristics of a model of a V/STOL airplane. The single-engine four-jet turbofan power plant of the airplane was simulated by inducing tunnel airflow through two large side inlets and injecting the decomposition products of hydrogen peroxide into the internal flow. The heated gas mixture was exhausted through four nozzles located on the sides of the fuselage under the wing, two near the wing leading edge and two forward of the trailing edge; the nozzles were deflected downward 1.5 deg and outward 5.0 deg to simulate cruise conditions. The wing of the model was a clipped delta with leading-edge sweep of 40 deg, aspect ratio of 3.06, taper ratio of 0.218, thickness-chord ratio of 0.09 at the root and 0.07 at the tip, and 10 deg negative dihedral. Aerodynamic and longitudinal stability coefficients were obtained for the model with the tail removed, and for horizontal-tail incidences of 0 deg and -5 deg. Data were obtained at Mach numbers from 0.60 to 1.00, angles of attack from 0 deg to 12 deg, and with jet total-pressure ratios up to 3.1. Jet operation generally caused a decrease in lift, an increase in pitching-moment coefficient, and a decrease in longitudinal stability at subsonic speeds. The jet interference effects on drag were detrimental at a Mach number of 0.60 and favorable at higher speeds for cruising-flight attitudes.

Author

Wind Tunnel Tests; Exhaust Gases; Dynamic Characteristics; Interference Drag; Aerodynamic Interference; Deflection

19980231071 NASA Langley Research Center, Hampton, VA USA

Aerodynamic Loading Characteristics at Mach Numbers from 0.80 to 1.20 of a 1/10-Scale Three-Stage Scout Model

Kelly, Thomas C., NASA Langley Research Center, USA; Sep. 1961; 164p; In English

Report No.(s): NASA-TN-D-945; L-1607; No Copyright; Avail: CASI; A08, Hardcopy; A02, Microfiche

Aerodynamic loads results have been obtained in the Langley 8-foot transonic pressure tunnel at Mach numbers from 0.80 to 1.20 for a 1/10-scale model of the upper three stages of the Scout vehicle. Tests were conducted through an angle-of-attack range from -8 deg to 8 deg at an average test Reynolds number per foot of about 4.0×10^6 . Results indicated that the peak negative pressures associated with expansion corners at the nose and transition flare exhibit sizeable variations which occur over a relatively small Mach number range. The magnitude of the variations may cause the critical local loading condition for the full-scale vehicle to occur at a Mach number considerably lower than that at which the maximum dynamic pressure occurs in flight. The addition of protuberances simulating antennas and wiring conduits had slight, localized effects. The lift carryover from the nose and transition flare on the cylindrical portions of the model generally increased with an increase in Mach number.

Author

Wind Tunnel Tests; Aerodynamic Loads; Aerodynamic Characteristics; Scale Models; Military Vehicles

03
AIR TRANSPORTATION AND SAFETY

Includes passenger and cargo air transport operations; and aircraft accidents.

19980227860 NASA Langley Research Center, Hampton, VA USA

Ditching Investigation of a Dynamic Model of a HU2K-1 Helicopter

Thompson, William C., NASA Langley Research Center, USA; 1961; 24p; In English

Report No.(s): NASA-TM-SX-626; NASA-AD-3142; N-AM-42; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Various configurations and approach conditions were investigated in order to determine the ditching behavior and the best ditching procedure. When ditchings were made without the float bags, the model rolled over on its side; when ditchings were made with the float bags inflated, the model remained upright. Late-flare and early-flare ditchings gave the same general behavior. Slight damage to the bottom surface of the scale-strength fuselage resulted for all test conditions.

Author

UH-2 Helicopter; Dynamic Models; Fuselages

19980228112 Sandia National Labs., Airworthiness Assurance NDI Validation Center, Albuquerque, NM USA

An Acoustic Emission Test for Aircraft Halon 1301 Fire Extinguisher Bottles *Final Report*

Beattie, A. G.; Apr. 1998; 20p; In English

Contract(s)/Grant(s): DTFA03-95-X-90002

Report No.(s): AD-A350935; DOT/FAA/AR-97/9; DOT/FAA/AAR,XH-433; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An acoustic emission test for aircraft Halon 1301 bottles has been developed, a prototype acoustic emission test system constructed, and over 200 used bottles tested at the repair facilities of the two manufacturers of these bottles. The system monitors a bottle with six acoustic sensors while the pressure of the bottle is raised by heating it in an oven. The sensors are held in position, with a fixed relationship between them, by a special fixture. This fixture is designed to fit spheres with diameters between 5 and 16 inches. Results of the tests on used bottles indicate that over 95 percent of the bottles showed no indication of significant defects. The rest had some indication of flaws or corrosion. However, all bottles tested to date have passed the hydrostatic test required by the U.S. Department of Transportation. Based upon this data, the Air Transport Association (ATA) requested an exemption from the DOT to allow their members to use this acoustic emission test in place of the hydrostatic test. This exemption, DOT - E 11850, was granted to the ATA on December 11, 1997.

DTIC

Fire Extinguishers; Commercial Aircraft; Nondestructive Tests; Acoustics; Fires; Bottles; Acoustic Emission

19980228242 NASA Lewis Research Center, Cleveland, OH USA

Crash-Fire Protection System for T-56 Turbopropeller Engine Using Water as Cooling and Inerting Agent

Busch, Arthur M., NASA Lewis Research Center, USA; Campbell, John A., NASA Lewis Research Center, USA; May 1959; 40p; In English

Report No.(s): NASA-MEMO-6-12-59E; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A crash-fire protection system to suppress the ignition of crash-spilled fuel that may be ingested by a T-56 turbopropeller engine is described. This system includes means for rapidly extinguishing the combustor flame and means for cooling and inerting with water the hot engine parts likely to ignite engine-ingested fuel. Combustion-chamber flames were extinguished in 0.07 second at the engine fuel manifold. Hot engine parts were inerted and cooled by 52 pounds of water discharged at ten engine stations. Performance trials of the crash-fire prevention system were conducted by bringing the engine up to takeoff temperature, stopping the normal fuel flow to the engine, starting the water discharge, and then spraying fuel into the engine to simulate crash-ingested fuel. No fires occurred during these trials, although fuel was sprayed into the engine from 0.3 second to 15 minutes after actuating the crash-fire protection system.

Author

Crashes; Fire Prevention; Combustion Chambers; Ignition; Stopping

19980228311 NASA Dryden Flight Research Facility, Edwards, CA USA

Flight Studies of Problems Pertinent to High-Speed Operation of Jet Transports

Butchart, Stanley P., NASA Dryden Flight Research Facility, USA; Fischel, Jack, NASA Dryden Flight Research Facility, USA; Tremant, Robert A., NASA Dryden Flight Research Facility, USA; Robinson, Glenn H., NASA Dryden Flight Research Facility, USA; Apr. 1959; 22p; In English

Report No.(s): NASA-MEMO-3-2-59H; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A flight investigation was made to assess the potential operational problems of jet transports in the transonic cruise range. In this study a large multiengine jet airplane having geometric characteristics fairly representative of the jet transport was used; however, in order to ensure general applicability of the results, the aerodynamic characteristics of the test airplane were varied to simulate a variety of jet- transport airplanes. Some of the specific areas investigated include: (1) an overall evaluation of longitudinal stability and control characteristics at transonic speeds, with an assessment of pitch-up characteristics, (2) the effect of buffeting on airplane operational speeds and maneuvering, (3) the desirable lateral-directional damping characteristics, (4) the desirable lateral-control characteristics, (5) an assessment of over-speed and speed-spread requirements, including the upset maneuver, and (6) an assessment of techniques and airplane characteristics for rapid descent and slow-down. The results presented include pilots' evaluation of the various problem areas and specific recommendations for possible improvement of jet-transport operations in the cruising speed range.

Author

Aerodynamic Characteristics; Jet Aircraft; Operational Problems; Transonic Speed; Transport Aircraft; Multiengine Vehicles

19980228493 Federal Aviation Administration, Washington, DC USA

Notices to Airmen: Domestic/International

Jun. 18, 1998; 236p; In English

Report No.(s): PB98-174170; Publ-ATA-10; No Copyright; Avail: CASI; A11, Hardcopy; A03, Microfiche

Table of Contents: Airway Notams; Airports, Facilities, and Procedural Notams; Airports, Facilities, and Procedural Notams; General FDC Notams; Part 95 Revisions to Minimum En Route IFR Altitudes and Changeover Points; International Notices to Airmen; and Graphic Notices.

NTIS

Airports; Graphs (Charts); National Airspace System; Air Navigation

19980228496 National Transportation Safety Board, Office of Judges, Washington, DC USA

National Transportation Safety Board Transportation Initial Decisions and Orders and Board Opinions and Orders: Adopted and Issued during the Month of June 1998

Jun. 1998; 278p; In English

Report No.(s): PB98-916706; NTSB/IDBOO-98/06; No Copyright; Avail: CASI; A13, Hardcopy; A03, Microfiche

This publication contains all Judge Initial Decisions and Board Opinions and Orders in Safety Enforcement and Seaman Enforcement cases for June 1998.

NTIS

Safety Management; Decisions

19980228497 National Transportation Safety Board, Office of Judges, Washington, DC USA

National Transportation Safety Board Transportation Initial Decisions and Orders and Board Opinions and Orders: Adopted and Issued during the Month of July 1998

Jul. 1998; 182p; In English

Report No.(s): PB98-916707; NTSB/IDBOO-98/07; No Copyright; Avail: CASI; A09, Hardcopy; A02, Microfiche

This publication contains all Judge Initial Decisions and Board Opinions and Orders in Safety Enforcement and Seaman Enforcement Cases for July 1998.

NTIS

Safety Management; Decisions

19980228502 Federal Aviation Administration, Washington, DC USA

Notices to Airmen: Domestic/International

Aug. 13, 1998; 204p; In English

Report No.(s): PB98-174683; Publ-ATA-10; No Copyright; Avail: CASI; A10, Hardcopy; A03, Microfiche

Table of Contents: Airway Notams; Airports, Facilities, and Procedural Notams; General FDC Notams; Part 95 Revisions to Minimum En Route IFR Altitudes and Changeover Points; International Notices to Airmen; and Graphic Notices.

NTIS

Airports; Graphs (Charts); National Airspace System; Air Navigation

19980230641 General Accounting Office, Resources, Community and Economic Development Div., Washington, DC USA

Aviation Safety: FAA Has Not Fully Implemented Weather-Related Recommendations

Jun. 1998; 78p; In English

Report No.(s): PB98-167653; GAO/RCED-98-130; B-278184; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

To minimize the danger that hazardous weather presents to the aviation system, the Federal Aviation Administration (FAA), in conjunction with other federal agencies, collects and analyzes weather information and disseminates this information to the users of the aviation system. However, between 1995, and 1997, one report by the National Research Council (NRC) and two reports by an FAA advisory committee cited inadequate interagency coordination and a lack of clarity about the agency's role in aviation weather. The reports also recommended steps FAA could take to provide better weather information to aviation users. Concerned about FAA's efforts to reduce weather-related accidents, you asked us to examine the actions FAA has taken to address the recommendations raised by NRC and FAA's advisory committee. In this report, we discuss FAA's actions in four areas of concern raised by the three reports: (1) policy and leadership, (2) interagency coordination, (3) meeting different types of users' needs for weather information, and (4) the level of funding provided for weather activities.

NTIS

Congressional Reports; Flight Safety; Aircraft Safety; Weather

19980230666 Federal Aviation Administration, Aviation Security Research and Development Div., Atlantic City, NJ USA

Development and Validation Plan for a Screener Readiness Test

Neiderman, Eric C., Federal Aviation Administration, USA; Fobes, J. L., Federal Aviation Administration, USA; Aug. 1998; 22p; In English

Contract(s)/Grant(s): DTFA03-98-D-00010

Report No.(s): PB98-168727; DOT/FAA/AR-98/31; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This document describes a plan to develop and validate a reliable, non-biased, secure test for initial screener training which can be used as a measure of knowledge of the screener's role in threat detection and checkpoint operations before on-the-job training.

NTIS

Personnel Development; Education

19980230680 NASA Dryden Flight Research Center, Edwards, CA USA

Flight Studies of Problems Pertinent to Low-Speed Operation of Jet Transports

Fischel, Jack, NASA Dryden Flight Research Center, USA; Butchart, Stanley P., NASA Dryden Flight Research Center, USA; Robinson, Glenn H., NASA Dryden Flight Research Center, USA; Tremant, Robert A., NASA Dryden Flight Research Center, USA; Apr. 1959; 22p; In English

Report No.(s): NASA-MEMO-3-1-59H; H-103; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Flight studies have been made of the low-speed operational regime of jet transports in order to assess potential operating problems. The study was performed utilizing a large multiengine jet airplane having geometric characteristics fairly representative of the jet transports; however, to insure general applicability of the results, the aerodynamic characteristics of the test airplane were varied to simulate a variety of jet-transport airplanes. The specific areas investigated include those of the take-off and landing, and the relation of these maneuvers to the 1 g stall speed and stalling characteristics. The take-off studies included evaluation of the factors affecting the take-off speed and attitude, including the effects of premature rotation and of over-rotation on ground run required. The approach and landing studies pertained to such factors as: desirable lateral-directional damping characteristics; lateral-control requirements; space-positioning limitations during approach under VFR or IFR conditions and requirements for glide-path controls; and evaluation of factors affecting the pilot's choice of landing speeds. Specific recommendations and some indication of desirable characteristics for the jet transports are advanced to alleviate possible operational difficulties or to improve operational performance in the low-speed range.

Author

Aerodynamic Characteristics; Low Speed; Jet Aircraft; Landing Speed; Transport Aircraft

AIRCRAFT DESIGN, TESTING AND PERFORMANCE

Includes aircraft simulation technology.

19980227824 NASA Langley Research Center, Hampton, VA USA

Applications of Power Spectral Analysis Methods to Maneuver Loads Obtained on Jet Fighter Airplanes During Service Operations

Mayer, John P., NASA Langley Research Center, USA; Hamer, Harold A., NASA Langley Research Center, USA; May 1961; 52p; In English

Report No.(s): NASA-TN-D-902; L-1557; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Power spectral densities of normal load factor have been obtained for two service operational training flights of a Republic F-84G airplane and three service operational training flights of a North American F-86A airplane in order to indicate the load-factor frequency content and possible uses of power spectral methods in analyzing maneuver load data. It was determined that the maneuvering load-factor time histories appeared to be described by a truncated normal distribution. The power spectral densities obtained were relatively level at frequencies below 0.03 cycle per second and varied inversely with approximately the cube of the frequency at the higher frequencies. In general, the frequency content was very low above 0.2 cycle per second. The load-factor peak distributions were estimated fairly well from the spectrum analysis. In addition, peak load data obtained during service operations of fighter-type airplanes with flight time totaling about 24,000 hours were examined and appeared to agree reasonably well with the type of equations obtained from spectrum peak-load distributions.

Author

Fighter Aircraft; Normal Density Functions; Spectrum Analysis; Education; Estimating; Flight Time; Frequencies

19980227826 NASA Langley Research Center, Hampton, VA USA

Methods for Determining the Optimum Design of Structures Protected from Aerodynamic Heating and Application to Typical Boost-Glide or Reentry Flight Paths

Harris, Robert S., Jr., NASA Langley Research Center, USA; Davidson, John R., NASA Langley Research Center, USA; Mar. 1962; 38p; In English

Report No.(s): NASA-TN-D-990; L-991; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

General equations are developed for the design of efficient structures protected from thermal environments typical of those encountered in boost-glide or atmospheric-reentry conditions. The method is applied to insulated heat-sink stressed-skin structures and to internally cooled insulated structures. Plates loaded in compression are treated in detail. Under limited conditions of plate buckling, high loading, and short flight periods, and for aluminum structures only, the weights of both configurations are nearly equal. Load parameters are found and are similar to those derived in previous investigations for the restricted case of a constant equilibrium temperature at the outside surface of the insulation.

Author

Boostglide Vehicles; Aerodynamic Heating; Thermal Environments; Stressed-Skin Structures; Flight Paths; Atmospheric Entry

19980227870 NASA Dryden Flight Research Facility, Edwards, CA USA

Flight Behavior of the X-2 Research Airplane to a Mach Number of 3.20 and a Geometric Altitude of 126,200 Feet

Day, Richard E., NASA Dryden Flight Research Facility, USA; Reisert, Donald, NASA Dryden Flight Research Facility, USA; Sep. 1959; 20p; In English

Report No.(s): NASA-TM-X-137; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The maximum Mach number and altitude capabilities of the Bell X-2 research airplane were achieved during a program conducted by the U.S. Air Force with Bell Aircraft Corp. providing operational support and the National Aeronautics and Space Administration providing instrumentation and advisory engineering assistance. A maximum geometric altitude of 126,200 feet was attained at a static pressure of 9.4 pounds per square foot and a dynamic pressure of 19.1 pounds per square foot. During the last flight of the airplane, a maximum Mach number of 3.20 was reached. The directionally divergent maneuver which terminated the final high Mach number flight was precipitated by the loss in directional stability that resulted from increasing the angle of attack. The yawing moment from the lateral control was sufficient to initiate the divergence and also to cause, indirectly, rolling moments that were greater than the aileron capabilities of the airplane. The ensuing violent motions-resulting from inertial roll coupling caused the loss of the aircraft.

Author

Flight Characteristics; Research Aircraft; Supersonic Speed; Directional Stability; Lateral Control; Aerodynamic Configurations

19980227972 NASA Langley Research Center, Hampton, VA USA

Hovering and Transition Flight Tests of a 1/5-Scale Model of a Jet-Powered Vertical-Attitude VTOL Research Airplane

Smith, Charles C., Jr., NASA Langley Research Center, USA; Dec. 1958; 32p; In English

Report No.(s): NASA-MEMO-1-10-27-58L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An experimental investigation has been made to determine the dynamic stability and control characteristics of a 1/5-scale flying model of a jet-powered vertical-attitude VTOL research airplane in hovering and transition flight. The model was powered with either a hydrogen peroxide rocket motor or a compressed-air jet exhausting through an ejector tube to simulate the turbojet engine of the airplane. The gyroscopic effects of the engine were simulated by a flywheel driven by compressed-air jets. In hovering flight the model was controlled by jet-reaction controls which consisted of a swiveling nozzle on the main jet and a movable nozzle on each wing tip; and in forward flight the model was controlled by elevons and a rudder. If the gyroscopic effects of the jet engine were not represented, the model could be flown satisfactorily in hovering flight without any automatic stabilization devices. When the gyroscopic effects of the jet engine were represented, however, the model could not be controlled without the aid of artificial stabilizing devices because of the gyroscopic coupling of the yawing and pitching motions. The use of pitch and yaw dampers made these motions completely stable and the model could then be controlled very easily. In the transition flight tests, which were performed only with the automatic pitch and yaw dampers operating, it was found that the transition was very easy to perform either with or without the engine gyroscopic effects simulated, although the model had a tendency to fly in a rolled and sideslipped attitude at angles of attack between approximately 25 and 45 deg because of static directional instability in this range.

Author

Hovering; Transition Flight; Flight Tests; Scale Models; Vertical Takeoff Aircraft; Dynamic Stability; Dynamic Control; Aircraft Performance; Control Systems Design

19980227992 NASA Langley Research Center, Hampton, VA USA

Rapid-Transition Tests of a 1/4-Scale Model of the VZ-2 Tilt-Wing Aircraft

Tosti, Louis P., NASA Langley Research Center, USA; Oct. 1961; 42p; In English

Report No.(s): NASA-TN-D-946; L-1683; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation of the longitudinal stability and control characteristics of a 1/4-scale model of the VZ-2 tilt-wing vertical-take-off- and-landing aircraft during rapid transitions has been made on the Langley control-line facility. Only the longitudinal characteristics were studied because with the control-line technique the other phases of the model motion are partially restrained. The rapid transitions from hovering to forward flight could be performed easily at any of the accelerations attempted; whereas, the transitions from forward flight to hovering were generally accompanied by a strong nose up pitching moment which at times was uncontrollable because of an inadequate amount of available pitch control. The model was more difficult to control during rapid decelerations than during slow decelerations and was also more difficult to control for rearward center-of-gravity conditions than for forward ones.

Author

Tilt Wing Aircraft; Vz-2 Aircraft; Scale Models; Horizontal Flight; Ground Based Control; Pitching Moments

19980227994 NASA Langley Research Center, Hampton, VA USA

An Investigation of Landing-Contact Conditions for Two Large Turbojet Transports and a Turboprop Transport During Routine Daylight Operations

Stickle, Joseph W., NASA Langley Research Center, USA; May 1961; 26p; In English

Report No.(s): NASA-TN-D-899; L-1528; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The National Aeronautics and Space Administration has recently completed a statistical investigation of landing-contact conditions for two large turbojet transports and a turboprop transport landing on a dry runway during routine daylight operations at the Los Angeles International Airport. Measurements were made to obtain vertical velocity, airspeed, rolling velocity, bank angle, and distance from the runway threshold, just prior to ground contact. The vertical velocities at touchdown for one of the turbojet airplanes measured in this investigation were essentially the same as those measured on the same type of airplane during a similar investigation (see NASA Technical Note D-527) conducted approximately 8 months earlier. Thus, it appeared that 8 months of additional pilot experience has had no noticeable tendency toward lowering the vertical velocities of this transport. Distributions of vertical velocities for the turbojet transports covered in this investigation were similar and considerably higher than those for the turboprop transport. The data for the turboprop transport were in good agreement with the data for the piston-engine transports (see NACA Report 1214 and NASA Technical Note D-147) for all the measured parameters. For the turbojet transports, 1 landing in 100 would be expected to equal or exceed a vertical velocity of approximately 4.2 ft/sec; whereas, for the turboprop transport, 1 landing in 100 would be expected to equal or exceed 3.2 ft/sec. The mean airspeeds at touchdown for the three transports ranged

from 22.5 percent to 26.6 percent above the stalling speed. Rolling velocities for the turbojet transports were considerably higher than those for the turboprop transport. Distributions of bank angles at contact for the three transports were similar. For each type of airplane, 1 landing in 100 would be expected to equal or exceed a bank angle at touchdown of approximately 3.0 deg. Distributions of touchdown distances for the three transports were also quite similar. Touchdown distances from the threshold for 1 landing in 100 ranged from 2,500 feet for the turboprop transport to 2,800 feet for one of the turbojet transports.

Author

Aircraft Landing; Turboprop Aircraft; Touchdown; Airspeed; Runways; Jet Aircraft

19980228059 NASA Langley Research Center, Hampton, VA USA

Effects of Canard Planform and Wing-Leading-Edge Modification on Low-Speed Longitudinal Aerodynamic Characteristics of a Canard Airplane Configuration

Spencer, Bernard, Jr., NASA Langley Research Center, USA; Aug. 1961; 52p; In English

Report No.(s): NASA-TN-D-958; L-1372; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An investigation has been conducted at low subsonic speeds to study the effects of canard planform and wing-leading-edge modification on the longitudinal aerodynamic characteristics of a general research canard airplane configuration. The basic wing of the model had a trapezoidal planform, an aspect ratio of 3.0, a taper ratio of 0.143, and an unswept 80-percent-chord line. Modifications to the wing included addition of full-span and partial-span leading-edge chord-extensions. Two canard planforms were employed in the study; one was a 60 deg sweptback delta planform and the other was a trapezoidal planform similar to that of the basic wing. Modifications to these canards included addition of a full-span leading-edge chord-extension to the trapezoidal planform and a fence to the delta planform. For the basic-wing-trapezoidal-canard configuration, rather abrupt increases in stability occurred at about 12 deg angle of attack. A slight pitch-up tendency occurred for the delta-canard configuration at approximately 8 deg angle of attack. A comparison of the longitudinal control effectiveness for the basic-wing-trapezoidal-canard combination and for the basic-wing-delta-canard combination indicates higher values of control effectiveness at low angles of attack for the trapezoidal canard. The control effectiveness for the delta-canard configuration, however, is seen to hold up for higher canard deflections and to higher angles of attack. Use of a full-span chord-extension deflected approximately 30 deg on the trapezoidal canard greatly improved the control characteristics of this configuration and enabled a sizeable increase in trim lift to be realized.

Author

Longitudinal Control; Canard Configurations; Aerodynamic Configurations; Aerodynamic Characteristics; Aspect Ratio; Leading Edges

19980228120 NASA Lewis Research Center, Cleveland, OH USA

Flight-Path Characteristics for Decelerating From Supercircular Speed

Luidens, Roger W., NASA Lewis Research Center, USA; Dec. 1961; 92p; In English

Report No.(s): NASA-TN-D-1091; E-1001; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

Characteristics of the following six flight paths for decelerating from a supercircular speed are developed in closed form: constant angle of attack, constant net acceleration, constant altitude" constant free-stream Reynolds number, and "modulated roll." The vehicles were required to remain in or near the atmosphere, and to stay within the aerodynamic capabilities of a vehicle with a maximum lift-drag ratio of 1.0 and within a maximum net acceleration G of 10 g's. The local Reynolds number for all the flight paths for a vehicle with a gross weight of 10,000 pounds and a 600 swept wing was found to be about 0.7×10^6 . With the assumption of a laminar boundary layer, the heating of the vehicle is studied as a function of type of flight path, initial G load, and initial velocity. The following heating parameters were considered: the distribution of the heating rate over the vehicle, the distribution of the heat per square foot over the vehicle, and the total heat input to the vehicle. The constant G load path at limiting G was found to give the lowest total heat input for a given initial velocity. For a vehicle with a maximum lift-drag ratio of 1.0 and a flight path with a maximum G of 10 g's, entry velocities of twice circular appear thermodynamically feasible, and entries at velocities of 2.8 times circular are aerodynamically possible. The predominant heating (about 85 percent) occurs at the leading edge of the vehicle. The total ablated weight for a 10,000-pound-gross-weight vehicle decelerating from an initial velocity of twice circular velocity is estimated to be 5 percent of gross weight. Modifying the constant G load flight path by a constant-angle-of-attack segment through a flight- to circular-velocity ratio of 1.0 gives essentially a "point landing" capability but also results in an increased total heat input to the vehicle.

Author

Flight Paths; Lift Drag Ratio; Angle of Attack; Deceleration; Free Flow; Laminar Boundary Layer

19980228146 NASA Langley Research Center, Hampton, VA USA

Flight Investigation of Effects of Transition, Landing Approaches, Partial-Power Vertical Descents, and Droop-Stop Pounding on the Bending and Torsional Moments Encountered by a Helicopter Rotor Blade

Ludi, LeRoy H., NASA Langley Research Center, USA; May 1959; 40p; In English

Report No.(s): NASA-MEMO-5-7-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Flight tests have been conducted with a single-rotor helicopter, one blade of which was equipped at 14 percent and 40 percent of the blade radius with strain gages calibrated to measure moments rather than stresses, to determine the effects of transition, landing approaches, and partial-power vertical descents on the rotor-blade bending and torsional moments. In addition, ground tests were conducted to determine the effects of static droop-stop pounding on the rotor-blade moments. The results indicate that partial-power vertical descents and landing approaches produce rotor-blade moments that are higher than the moments encountered in any other flight condition investigated to date with this equipment. Decelerating through the transition region in level flight was found to result in higher vibratory moments than accelerating through this region. Deliberately induced static droop-stop pounding produced flapwise bending moments at the 14-percent-radius station which were as high as the moments experienced in landing approaches and partial-power vertical descents.

Author

Flight Tests; Torsional Stress; Bending Moments; Vertical Landing

19980228159 NASA Langley Research Center, Hampton, VA USA

A Summary of Operating Conditions Experienced by Three Military Helicopters and a Mountain-Based Commercial Helicopter

Connor, Andrew B., NASA Langley Research Center, USA; Oct. 1960; 24p; In English

Report No.(s): NASA-TN-D-432; L-1157; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The results of a survey of the flight conditions experienced by three military helicopters engaged in simulated and actual military missions, and a commercial helicopter operated in the mountainous terrain surrounding Denver, CO, are presented. The data, obtained with NASA helicopter VGHN recorders, represent 813 flights or 359 flying hours, and are compared where applicable to previous survey results. The current survey results show that none of the helicopters exceeded the maximum design airspeed. One military helicopter, used for instrument flight training, never exceeded 70 percent of its maximum design airspeed. The rates of climb and descent utilized by the IFR training helicopter and of the mountain-based helicopter were generally narrowly distributed within all the airspeed ranges. The number of landings per hour for all four of the helicopters ranged from 1.6 to 3.3. The turbine-engine helicopter experienced more frequent normal-acceleration increments above a threshold of $\pm 0.4g$ (where g is acceleration due to gravity) than the mountain-based helicopter, but the mountain-based helicopter experienced acceleration increments of greater magnitude. Limited rotor rotational speed time histories showed that all the helicopters were operated at normal rotor speeds during all flight conditions.

Author

Military Helicopters; Flight Conditions; Climbing Flight; Airspeed; Descent; Military Operations

19980228163 NASA Dryden Flight Research Center, Edwards, CA USA

Full Flight Envelope Direct Thrust Measurement on a Supersonic Aircraft

Connors, Timothy R., NASA Dryden Flight Research Center, USA; Sims, Robert L., NASA Dryden Flight Research Center, USA; Jul. 1998; 34p; In English; 34th; Propulsion, 13-15 Jul. 1998, Cleveland, OH, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA

Contract(s)/Grant(s): RTOP 529-20-04-00-33

Report No.(s): NASA/TM-1998-206560; H-2266; NAS 1.15:206560; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Direct thrust measurement using strain gages offers advantages over analytically-based thrust calculation methods. For flight test applications, the direct measurement method typically uses a simpler sensor arrangement and minimal data processing compared to analytical techniques, which normally require costly engine modeling and multisensor arrangements throughout the engine. Conversely, direct thrust measurement has historically produced less than desirable accuracy because of difficulty in mounting and calibrating the strain gages and the inability to account for secondary forces that influence the thrust reading at the engine mounts. Consequently, the strain-gage technique has normally been used for simple engine arrangements and primarily in the subsonic speed range. This paper presents the results of a strain gage-based direct thrust-measurement technique developed by the NASA Dryden Flight Research Center and successfully applied to the full flight envelope of an F-15 aircraft powered by two F100-PW-229 turbofan engines. Measurements have been obtained at quasi-steady-state operating conditions at maximum non-augmented and maximum augmented power throughout the altitude range of the vehicle and to a maximum speed of Mach

2.0 and are compared against results from two analytically-based thrust calculation methods. The strain-gage installation and calibration processes are also described.

Author

F-15 Aircraft; Turbofan Engines; Supersonic Aircraft; Mach Number

19980228207 NASA Langley Research Center, Hampton, VA USA

Evaluation of the Levy Method as Applied to Vibrations of a 45 deg Delta Wing

Kruszewski, Edwin T., NASA Langley Research Center, USA; Waner, Paul G., Jr., NASA Langley Research Center, USA; Feb. 1959; 52p; In English

Report No.(s): NASA-MEMO-2-2-59L; L-153; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

The Levy method which deals with an idealized structure was used to obtain the natural modes and frequencies of a large-scale built-up 45 deg. delta wing. The results from this approach, both with and without the effects of transverse shear, were compared with the results obtained experimentally and also with those calculated by the Stein-Sanders method. From these comparisons it was concluded that the method as proposed by Levy gives excellent results for thin-skin delta wings, provided that corrections are made for the effect of transverse shear.

Author

Delta Wings; Aeronautical Engineering; Vibration; Transverse Loads; Thin Wings; Thickness Ratio; Thin Airfoils

19980228208 NASA Ames Research Center, Moffett Field, CA USA

An Experimental Investigation of a Triangular Wing of Aspect Ratio 2 and a Body Warped to be Trimmed at $M = 2.24$

Adams, Gaynor J., NASA Ames Research Center, USA; Boyd, John W., NASA Ames Research Center, USA; Feb. 1959; 42p; In English

Report No.(s): NASA-MEMO-2-3-59A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A cambered and twisted triangular wing of aspect ratio 2 in combination with a cambered body was investigated experimentally to determine the effectiveness of the camber in reducing the drag due to lift at trim at supersonic speeds. Four arrangements were tested comprising all combinations of a symmetrical and a cambered wing with a symmetrical and a cambered body. The camber shape investigated was derived by linearized lifting surface theory for triangular wings with sonic leading edges and satisfied the requirement that the wing be trimmed at the design Mach number and lift coefficient. The experimental results for the cambered wing and cambered body showed that the drag coefficient at trim was always greater, at the same lift coefficient, than that for the untrimmed symmetrical wing and body. The trim lift coefficient was positive and decreased with increasing Mach number. At the design Mach number of 2.24, the trim lift coefficient was somewhat lower and the drag coefficient was higher than values predicted by linearized lifting surface theory for the wing alone. A comparison of the trim lift-drag ratio of the cambered wing and cambered body with values obtained by trimming the symmetrical wing and symmetrical body either with a canard or a trailing-edge flap showed that, at approximately the design Mach number the cambered configuration developed a somewhat higher value than the trailing-edge flap configuration but a lower value than the canard configuration.

Author

Aerodynamic Coefficients; Twisted Wings; Supersonic Speed; Lift Drag Ratio; Drag Reduction; Delta Wings; Canard Configurations; Cambered Wings; Aspect Ratio; Aerodynamic Drag

19980228232 NASA Ames Research Center, Moffett Field, CA USA

Flight Measurements of the Effect of a Controllable Thrust Reverser on the Flight Characteristics of a Single-Engine Jet Airplane

Anderson, Seth B., NASA Ames Research Center, USA; Cooper, George E., NASA Ames Research Center, USA; Faye, Alan E., Jr., NASA Ames Research Center, USA; May 1959; 48p; In English

Report No.(s): NASA-MEMO-4-26-59A; A-135; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A flight investigation was undertaken to determine the effect of a fully controllable thrust reverser on the flight characteristics of a single-engine jet airplane. Tests were made using a cylindrical target-type reverser actuated by a hydraulic cylinder through a "beep-type" cockpit control mounted at the base of the throttle. The thrust reverser was evaluated as an in-flight decelerating device, as a flight path control and airspeed control in landing approach, and as a braking device during the ground roll. Full deflection of the reverser for one reverser configuration resulted in a reverse thrust ratio of as much as 85 percent, which at maximum engine power corresponded to a reversed thrust of 5100 pounds. Use of the reverser in landing approach made possible a wide selection of approach angles, a large reduction in approach speed at steep approach angles, improved control of flight path angle, and more accuracy in hitting a given touchdown point. The use of the reverser as a speed brake at lower airspeeds was compro-

misled by a longitudinal trim change. At the lower airspeeds and higher engine powers there was insufficient elevator power to overcome the nose-down trim change at full reverser deflection.

Author

Thrust Reversal; Flight Characteristics; Approach Control; Aircraft Brakes; Cylindrical Bodies; Aircraft Landing; Measuring Instruments; Jet Engines

19980228247 NASA Langley Research Center, Hampton, VA USA

Analytical and Experimental Determination of the Coupled Natural Frequencies and Mode Shapes of a Dynamic Model of a Single-Rotor Helicopter

Silveira, Milton A., NASA Langley Research Center, USA; Brooks, George W., NASA Langley Research Center, USA; Dec. 1958; 46p; In English

Report No.(s): NASA-MEMO-11-5-58L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A combined analytical and experimental determination is made of the coupled natural frequencies and mode shapes in the longitudinal plane of symmetry for a dynamic model of a single-rotor helicopter. The analytical phase is worked out on the basis of a seven-degree-of-freedom system combining elastic deflections of the rotor blades, rotor shaft, pylon, and fuselage. The calculated coupled frequencies are first compared with calculated uncoupled frequencies to show the general effects of coupling and then with measured coupled frequencies to determine the extent to which the coupled frequencies can be calculated. The coupled mode shapes are also calculated and were observed visually with stroboscopic lights during the tests. A comparison of the coupled and uncoupled natural frequencies shows that significant differences exist between these frequencies for some of the modes. Good agreement is obtained between the measured and calculated values for the coupled natural frequencies and mode shapes. The results show that the coupled natural frequencies and mode shapes can be determined by the analytical procedure presented herein with sufficient accuracy if the mass and stiffness distributions of the various components of the helicopter are known.

Author

Numerical Analysis; Data Acquisition; Experimentation; Coupled Modes; Frequencies; Shapers; Dynamic Models; Rotary Wings

19980228266 NASA Langley Research Center, Hampton, VA USA

Take-off Distances of a Supersonic Transport Configuration as Affected by Airplane Rotation During the Take-off Run

Hall, Albert W., NASA Langley Research Center, USA; Oct. 1961; 24p; In English

Report No.(s): NASA-TN-D-982; L-1728; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The take-off distances over a 35-foot obstacle have been determined for a supersonic transport configuration characterized by a low maximum lift coefficient at a high angle of attack and by high drag due to lift. These distances were determined analytically by means of an electronic digital computer. The effects of rotation speed, rotation angle, and rotation time were determined. A few configuration changes were made to determine the effects of thrust-weight ratio, wing loading, maximum lift coefficient, and induced drag on the take-off distance. The required runway lengths based on Special Civil Air Regulation No. SR-422B were determined for various values of rotation speed and compared with those based on full engine power. Increasing or decreasing the rotation speed as much as 5 knots from the value at which the minimum take-off distance occurred increased the distance only slightly more than 1 percent for the configuration studied. Under-rotation by 1 deg to 1.5 deg increased the take-off distance by 9 to 15 percent. Increasing the time required for rotation from 3 to 5 seconds had a rather small effect on the take-off distance when the values of rotation speed were near the values which result in the shortest take-off distance. When the runway length is based on full engine power rather than on SR-422B, the rotation speed which results in the shortest required runway length is 10 knots lower and the runway length is 4.3 percent less.

Author

Takeoff; Supersonic Transports; Aerodynamic Coefficients; Runways; Induced Drag

19980228279 NASA, Washington, DC USA

Response of a Helicopter Rotor to an Increase in Collective Pitch for the Case of Vertical Flight *Reponse d'un Rotor d'Helicoptere a une Augmentation du pas General dans le Cas du Vol Vertical*

Redont, Jean; Valensi, Jacques; Soulez-Lariviere, Jean; Technique et Science Aeronautiques; Jan. 1961, No. 3, pp. 177-183; In English

Report No.(s): NASA-TT-F-55; L-1344; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A theoretical analysis, considering unsteady effects, shows that the response to collective pitch while in descending flight is largely dependent upon the rapidity of control application. Experimental tests confirm the theoretical results.

Author

Rotary Wings; Vertical Flight; Pitch (Inclination)

19980228286 NASA Ames Research Center, Moffett Field, CA USA

STOL Characteristics of a Propeller-Driven, Aspect-Ratio-10, Straight-Wing Airplane with Boundary-Layer Control Flaps, as Estimated from Large-Scale Wind-Tunnel Tests

Weiberg, James A, NASA Ames Research Center, USA; Holzhauser, Curt A., NASA Ames Research Center, USA; Jun. 1961; 60p; In English

Report No.(s): NASA-TN-D-1032; A-423; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

A study is presented of the improvements in take-off and landing distances possible with a conventional propeller-driven transport-type airplane when the available lift is increased by propeller slipstream effects and by very effective trailing-edge flaps and ailerons. This study is based on wind-tunnel tests of a 45-foot span, powered model, with BLC on the trailing-edge flaps and controls. The data were applied to an assumed airplane with four propellers and a wing loading of 50 pounds per square foot. Also included is an examination of the stability and control problems that may result in the landing and take-off speed range of such a vehicle. The results indicated that the landing and take-off distances could be more than halved by the use of highly effective flaps in combination with large amounts of engine power to augment lift (STOL). At the lowest speeds considered (about 50 knots), adequate longitudinal stability was obtained but the lateral and directional stability were unsatisfactory. At these low speeds, the conventional aerodynamic control surfaces may not be able to cope with the forces and moments produced by symmetric, as well as asymmetric, engine operation. This problem was alleviated by BLC applied to the control surfaces.

Author

Short Takeoff Aircraft; Boundary Layer Control; Flaps (Control Surfaces); Lateral Stability; Trailing Edge Flaps; Aerodynamics

19980228289 NASA Langley Research Center, Hampton, VA USA

Effect of Six Missile-Bay Baffle Configurations and a Rocket End Plate on Ejection Releases of an MB-1 Rocket from a 0.05 Scale Model of the Convair F-106A Airplane

Hinson, William F., NASA Langley Research Center, USA; Lee, John B., NASA Langley Research Center, USA; 1959; 52p; In English

Report No.(s): NASA-MEMO-4-29-59L; AF-AM-57; L-361; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

As a continuation of an investigation of the release characteristics of an MB-1 rocket carried internally by the Convair F-106A airplane, six missile-bay baffle configurations and a rocket end plate have been investigated in the 27- by 27-inch preflight jet of the NASA Wallops Station. The MB-1 rocket used had retractable fins and was ejected from a missile bay modified by the addition of six different baffle configurations. For some tests a rocket end plate was added to the model. Dynamically scaled models (0.04956 scale) were tested at a simulated altitude of 22,450 feet and Mach numbers of 0.86, 1.59, and 1.98, and at a simulated altitude of 29,450 feet and a Mach number of 1.98. The results of this investigation indicate that the missile-bay baffle configurations and the rocket end plate may be used to reduce the positive pitch amplitude of the MB-1 rocket after release. The initial negative pitching velocity applied to the MB-1 rocket might then be reduced in order to maintain a near-level-flight attitude after release. As the fuselage angle of attack is increased, the negative pitch amplitude of the rocket is decreased.

Author

F-106 Aircraft; Baffles; Ejection; End Plates; Scale Models; Missile Configurations; Altitude Simulation

19980228302 NASA Langley Research Center, Hampton, VA USA

Flight Tests of A 1/8-Scale Model of the Bell D-188A Jet VTOL Airplane

Smith, Charles C., Jr., NASA Langley Research Center, USA; Sep. 30, 1971; 30p; In English

Report No.(s): NASA-MEMO-3-16-59L; TED-AD-3147; L-241; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The Bell D-188A VTOL airplane is a horizontal-attitude VTOL fighter with tilting engine nacelles at the tips of a low-aspect-ratio unswept wing and additional engines in the fuselage. The model could be flown smoothly in hovering and transition flight. In forward flight the model could be flown smoothly at the lower angles of attack but experienced an uncontrollable directional divergence at angles of attack above about 16 deg.

Author

Vertical Takeoff Aircraft; Flight Tests; Aircraft Configurations; Unswept Wings; Scale Models; Engine Design; Tilted Propellers; Engine Airframe Integration; Low Aspect Ratio Wings

19980228303 NASA Langley Research Center, Hampton, VA USA

Comparison of Measured Flapwise Structural Bending Moments on a Teetering Rotor Blade With Results Calculated From the Measured Pressure Distribution

Mayo, Alton P., NASA Langley Research Center, USA; Mar. 1959; 34p; In English

Report No.(s): NASA-MEMO-2-28-59L; L-140; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Flapwise bending moments were calculated for a teetering rotor blade using a reasonably rapid theoretical method in which airloads obtained from wind-tunnel tests were employed. The calculated moments agreed reasonably well with those measured with strain gages under the same test conditions. The range of the tests included one hovering and two forward-flight conditions. The rotor speed for the test was very near blade resonance, and difficult-to-calculate resonance effects apparently were responsible for the largest differences between the calculated and measured harmonic components of blade bending moments. These differences, moreover, were largely nullified when the harmonic components were combined to give a comparison of the calculated and measured blade total- moment time histories. The degree of agreement shown is therefore considered adequate to warrant the use of the theoretical method in establishing and applying methods of prediction of rotor-blade fatigue loads. At the same time, the validity of the experimental methods of obtaining both airload and blade stress measurement is also indicated to be adequate for use in establishing improved methods for prediction of rotor-blade fatigue loads during the design stage. The blade stiffnesses and natural frequencies were measured and found to be in close agreement with calculated values; however, for a condition of blade resonance the use of the experimental stiffness values resulted in better agreement between calculated and measured blade stresses.

Author

Bending Moments; Aerodynamic Loads; Stress Measurement; Wind Tunnel Tests; Teetering; Structural Analysis; Flaps (Control Surfaces); Rotary Wings

19980228312 NASA Dryden Flight Research Center, Edwards, CA USA

Roll Utilization of an F-100A Airplane During Service Operational Flying

Matranga, Gene J., NASA Dryden Flight Research Center, USA; Jan. 1959; 50p; In English

Report No.(s): NASA-MEMO-12-1-58H; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

As a means of evaluating the roll utilization of a fighter airplane capable of supersonic speeds, an instrumented North American F-100A fighter airplane was flown by U.S. Air Force pilots at Nellis Air Force Base, NV, during 20 hours of service operational flying. Mach numbers up to 1.22 and altitudes up to 50,000 feet were realized in this investigation. Results of the study showed that except for high g barrel rolls performed as evasive maneuvers and rolls performed in acrobatic flying, rolling was utilized primarily as a means of changing heading. Acrobatic and air combat maneuvering produced the largest bank angles (1,200 deg), roll velocities (3.3 radians/sec), rolling accelerations (8 radians/sq sec) and sideslip angles (10.8 deg). Full aileron deflections were utilized on numerous occasions. Although high rolling velocities and accelerations also were experienced during several air-to-air gunnery missions, generally, air-to-air gunnery and air-to-ground gunnery and bombing required only two-thirds of maximum aileron deflection. The air-to-air gunnery and air combat maneuvers initiated from supersonic speeds utilized up to two-thirds aileron deflection and bank angles of less than 18 deg and resulted in rolling velocities and accelerations of 2 radians per second and 4.6 radians/sq sec, respectively. Rolling maneuvers were often initiated from high levels of normal acceleration, but from levels of negative normal acceleration only once.

Author

F-100 Aircraft; Roll; High Acceleration; Supersonic Speed

19980228318 NASA Langley Research Center, Hampton, VA USA

Study of Taxiing Problems Associated with Runway Roughness

Milwitzky, Benjamin, NASA Langley Research Center, USA; Mar. 1959; 14p; In English

Report No.(s): NASA-MEMO-2-21-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This paper briefly summarizes available statistical data on airplane taxi operations, examines the profiles and power spectra of four selected runways and taxiways covering a wide range of surface roughness, considers (on the basis of theoretical and experimental results) the loads resulting from taxiing on such runways over a range of speeds and, by synthesis of the aforementioned results, proposes new criteria for runway and taxiway smoothness which are applicable to new construction and may also be used as a guide for determining when repairs are necessary.

Author

Runways; Surface Roughness; Taxiing; Runway Conditions

19980228320 NASA Langley Research Center, Hampton, VA USA

A Limited Study of a Hypothetical Winged Anti-ICBM Point-Defense Missile

Brown, Clarence A., Jr., NASA Langley Research Center, USA; Edwards, Frederick G., NASA Langley Research Center, USA; Jun. 1959; 36p; In English

Report No.(s): NASA-MEMO-2-14-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A preliminary investigation was conducted to determine whether a warhead stage of an antimissile missile could be placed within an arbitrary 2-nautical-mile-radius maneuver cylinder around an intercontinental-ballistic-missile (ICBM) flight path above an altitude of 140,000 feet, a horizontal range of 40 nautical miles, at a flight-path angle of approximately 20 deg, and within 50 seconds after take-off using only aerodynamic forces to turn the antimissile missile. The preliminary investigation indicated that an antimissile missile using aerodynamic forces for turning was capable of intercepting the ICBM for the stated conditions of this study although the turning must be completed below an altitude of approximately 70,000 feet to insure that the antimissile missile will be at the desired flight-path angle. Trim lift coefficients on the order of 2 to 3 and a maximum normal-acceleration force of from 25g to 35g were necessary to place the warhead stage in intercept position. The preliminary investigation indicated that for the two boosters investigated the booster having a burning time of 10 seconds gave greater range up the ICBM flight path than did the booster having a burning time of 15 seconds for the same trim lift coefficient and required the least trim lift coefficient for the same range.

Author

Antimissile Missiles; Intercontinental Ballistic Missiles; Flight Paths; Aerodynamic Forces; Aerodynamic Coefficients; Burning Time

19980228323 National Advisory Committee for Aeronautics. Ames Aeronautical Lab., Moffett Field, CA USA

The Effect of Lower Surface Spoilers on the Transonic Trim Change of a Wind-Tunnel Model of a Fighter Airplane Having a Modified Delta Wing

Robinson, Robert C., National Advisory Committee for Aeronautics. Ames Aeronautical Lab., USA; Feb. 1959; 54p; In English
Report No.(s): NASA-MEMO-12-27-58A; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

In an attempt to find an aerodynamic means of counteracting the transonic trim change of a fighter airplane, lower surface spoilers were tested on a 0.055-scale wind-tunnel model. The Mach number range of the tests was 0.8 to 1.2 at Reynolds numbers of approximately 4 million. Although the spoilers produced a moderate decrease in the trim change at low altitudes, they also produced a large increase in drag. Pressure-distribution tests with external fuel tanks showed large pressure changes on the lower surface of the wing due to the tanks.

Author

Wind Tunnel Tests; Fighter Aircraft; Transonic Speed; Scale Models; Aerodynamic Balance; Aerodynamic Characteristics; Delta Wings

19980228350 NASA Dryden Flight Research Center, Edwards, CA USA

Aerodynamic and Landing Measurements Obtained During the First Powered Flight of the North American X-15 Research Airplane

Jan. 1960; 42p; In English

Report No.(s): NASA-TM-X-269; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

During the first powered flight of the North American X-15 research airplane on September 17, 1959, a Mach number of 2.1 and an altitude of 52,000 feet were attained. Static and dynamic maneuvers were performed to evaluate the characteristics of the airplane at subsonic and supersonic speeds. Data from these maneuvers as well as from the launch and landing phases are presented, discussed, and compared with predicted values. The rate of separation of the X-15 from the B-52 carrier airplane at launch was less than that predicted by wind-tunnel studies and was less rapid than in the lightweight condition of the initial glide flight. In addition, the angular motions and bank angle attained following the launch were of lesser magnitude than in the glide flight. Stable longitudinal-stability trends were apparent during the acceleration to maximum speed, and the pilot reported experiencing little or no transonic trim excursions. An inexplicable high-frequency vibration, which occurred at Mach numbers above 1.4, is being investigated further. Essentially linear lift and stability characteristics were indicated within the limited ranges of angle of attack and angle of sideslip investigated. The dynamic longitudinal and lateral-directional stability and control-effectiveness characteristics appeared satisfactory to the pilot. Although the longitudinal- and lateral-directional-damping ratios showed no significant change from subsonic to supersonic speeds, on the basis of time to damp, the damping characteristics at supersonic speeds appeared to the pilot to be somewhat improved over those at subsonic speeds.

Author

B-52 Aircraft; Directional Stability; Launching; Controllability; Angular Velocity

19980228371 NASA Langley Research Center, Hampton, VA USA

Some Static Oscillatory and Free Body Tests of Blunt Bodies at Low Subsonic Speeds

Lichtenstein, Jacob H., NASA Langley Research Center, USA; Fisher, Lewis R., NASA Langley Research Center, USA; Scher, Stanley H., NASA Langley Research Center, USA; Lawrence, George F., NASA Langley Research Center, USA; Apr. 1959; 32p; In English

Report No.(s): NASA-MEMO-2-22-59L; L-157; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Some blunt-body shapes considered suitable for entry into the earth's atmosphere were tested by both static and oscillatory methods in the Langley stability tunnel. In addition, free-fall tests of some similar models were made in the Langley 20-foot free-spinning tunnel. The results of the tests show that increasing the flare of the body shape increased the dynamic stability and that for flat-faced shapes increasing the corner radius increased the stability. The test data from the Langley stability tunnel were used to compute the damping factor for the models tested in the Langley 20-foot free-spinning tunnel. For these cases in which the damping factor was low, $-1/2$ or less, the stability was critical and sensitive to disturbance. When the damping factor was about -2 , damping was generally obtained.

Author

Blunt Bodies; Atmospheric Entry; Subsonic Speed; Dynamic Stability; Stability Tests; Wind Tunnel Tests; Damping

19980228395 NASA Langley Research Center, Hampton, VA USA

Response of a WB-47E Airplane to Runway Roughness at Eielson AFB, Alaska, September 1964

Morris, Garland J., NASA Langley Research Center, USA; Hall, Albert W., NASA Langley Research Center, USA; Mar. 1965; 26p; In English

Report No.(s): NASA-TM-SX-1076; L-4439; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been conducted to measure the response of a WB-47E airplane to the roughness of the runway at Eielson AFB, Alaska. The acceleration level in the pilot's compartment and the pitching oscillation of the airplane were found to be sufficiently high to possibly cause pilot discomfort and have an adverse effect on the precision of take-off.

Author

Runways; Surface Roughness; Runway Conditions

19980228448 NASA Langley Research Center, Hampton, VA USA

Analytical Method of Approximating the Motion of a Spinning Vehicle with Variable Mass and Inertia Properties Acted Upon by Several Disturbing Parameters

Buglia, James J., NASA Langley Research Center, USA; Young, George R., NASA Langley Research Center, USA; Timmons, Jesse D., NASA Langley Research Center, USA; Brinkworth, Helen S., NASA Langley Research Center, USA; 1961; 24p; In English

Report No.(s): NASA-TR-R-110; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An analytical method has been developed which approximates the dispersion of a spinning symmetrical body in a vacuum, with time-varying mass and inertia characteristics, under the action of several external disturbances-initial pitching rate, thrust misalignment, and dynamic unbalance. The ratio of the roll inertia to the pitch or yaw inertia is assumed constant. Spin was found to be very effective in reducing the dispersion due to an initial pitch rate or thrust misalignment, but was completely ineffective in reducing the dispersion of a dynamically unbalanced body.

Author

Approximation; Symmetrical Bodies; Misalignment; Inertia

19980228450 NASA Langley Research Center, Hampton, VA USA

Hydrodynamic and Aerodynamic Characteristics of a Model of a Supersonic Multijet Water-Based Aircraft Equipped with Supercavitating Hydrofoils

McKann, Robert E., NASA Langley Research Center, USA; Blanchard, Ulysse J., NASA Langley Research Center, USA; Pearson, Albin O., NASA Langley Research Center, USA; Feb. 1960; 48p; In English

Report No.(s): NASA-TM-X-191; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The hydrodynamic and aerodynamic characteristics of a model of a multijet water-based Mach 2.0 aircraft equipped with hydrofoils have been determined. Takeoff stability and spray characteristics were very good, and sufficient excess thrust was available for takeoff in approximately 32 seconds and 4,700 feet at a gross weight of 225,000 pounds. Longitudinal and lateral stability during smooth-water landings were good. Lateral stability was good during rough-water landings, but forward location of the hydrofoils or added pitch damping was required to prevent diving. Hydrofoils were found to increase the aerodynamic lift-curve slope and to increase the aerodynamic drag coefficient in the transonic speed range, and the maximum lift-drag ratio decreased

from 7.6 to 7.2 at the cruise Mach number of 0.9. The hydrofoils provided an increment of positive pitching moment over the Mach number range of the tests (0.6 to 1.42) and reduced the effective dihedral and directional stability.

Author

Aerodynamic Characteristics; Lift Drag Ratio; Longitudinal Stability; Directional Stability; Aerodynamic Drag; Aerodynamic Coefficients

19980228470 Massachusetts Inst. of Tech., Cambridge, MA USA

Effect of Cascade Parameters on Rotating Stall

Stenning, A. H., Massachusetts Inst. of Tech., USA; Seidel, B. S., Massachusetts Inst. of Tech., USA; Senoo, Y., Massachusetts Inst. of Tech., USA; Apr. 1959; 36p; In English

Report No.(s): NASA-MEMO-3-16-59W; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Analysis of the vortex model proposed by Kriebel, Seidel, and Schwind shows this representation of rotating stall satisfies, at least approximately, the requirements at the cascade. Cascade-parameter-variation effects on rotating stall were studied in a circular cascade and single-stage compressor. Modification of the single-stage compressor stopped the rotating-stall pattern and permitted observation of the pressure and velocity distribution around the annulus. Closer observation might be possible with proper flow-visualization techniques, such as a water pump.

Author

Rotating Stalls; Flow Visualization; Velocity Distribution; Pressure Distribution

19980230604 NASA Langley Research Center, Hampton, VA USA

Evaluation of Several Approximate Methods for Calculating the Symmetrical Bending-Moment Response of Flexible Airplanes to Isotropic Atmospheric Turbulence

Bennett, Floyd V., NASA Langley Research Center, USA; Yntema, Robert T., NASA Langley Research Center, USA; Mar. 1959; 58p; In English

Report No.(s): NASA-MEMO-2-18-59L; L-143; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Several approximate procedures for calculating the bending-moment response of flexible airplanes to continuous isotropic turbulence are presented and evaluated. The modal methods (the mode-displacement and force-summation methods) and a matrix method (segmented-wing method) are considered. These approximate procedures are applied to a simplified airplane for which an exact solution to the equation of motion can be obtained. The simplified airplane consists of a uniform beam with a concentrated fuselage mass at the center. Airplane motions are limited to vertical rigid-body translation and symmetrical wing bending deflections. Output power spectra of wing bending moments based on the exact transfer-function solutions are used as a basis for the evaluation of the approximate methods. It is shown that the force-summation and the matrix methods give satisfactory accuracy and that the mode-displacement method gives unsatisfactory accuracy.

Author

Evaluation; Procedures; Computation; Approximation; Bending Moments; Rigid Structures

19980230614 Aix-Marseille Univ., Inst. of Fluid Mechanics, Marseille, France

Wind-Tunnel Study of the Response in Lift of a Rotor to an Increase in Collective Pitch in the Case of Vertical Flight Near the Autorotative Regime *Etude en soufflerie de la reponse de la portance d'un rotor a une augmentation de pas general, dans le cas du vol vertical de descente a un regime voisin de l'autorotation*

Rebont, Jean, Aix-Marseille Univ., France; Valensi, Jacques; Soulez-Lariviere, Jean, Aix-Marseille Univ., France; Comptes Rendus; Apr. 1960; Volume 247, No. 9, pp. 738; In English

Report No.(s): NASA-TT-F-17; L-454; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

It has been shown by the calculations of a preceding note that the effect on the lift of a rotor due to an increase in the effective collective pitch, while in the course of steady descending vertical flight near the autorotative regime, depends essentially upon the speed of the pitch change. Experiment confirms the result.

Author

Wind Tunnel Tests; Rotors; Lift; Rotor Aerodynamics; Pitch (Inclination)

19980230622 NASA, Washington, DC USA

Optimum Airplane Flight Paths *Le Evoluzioni Ottime di un Aereo*

Cicala, Placido; Atti della Accademia delle Scienze di Torino; Oct. 1959; Volume 89, pp. 350-358; In English; Translated by R. H. Cramer, Johns Hopkins Univ., Silver Springs, MD

Report No.(s): NASA-TT-F-4; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The generalized equations are discussed which pertain to an airplane executing its flight path in a single vertical plane, under a sequence of stipulations specifying the nature of the local optima which are distinctly different in character along successive portions of this path. The aerodynamic and propulsive characteristics of the airplane are allowed to be specified with complete generality.

Author

Flight Paths; Differential Equations; Optimization

19980230623 NASA Langley Research Center, Hampton, VA USA

Application of the Method of Stein and Sanders to the Calculation of Vibration Characteristics of a 45 deg Delta-Wing Specimen

Hedgepeth, John M., NASA Langley Research Center, USA; Warner, Paul G., Jr., NASA Langley Research Center, USA; Feb. 1959; 30p; In English

Report No.(s): NASA-MEMO-2-1-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Generalized influence coefficients are calculated by the method of NACA TN 3640 for a large-scale, built-up, 450 delta-wing specimen. These are used together with appropriate generalized masses to obtain the natural modes and frequencies in symmetric and antisymmetric free-free vibration. The resulting frequencies are compared with those obtained experimentally and are found to be consistently high. Possible sources of the disparities are discussed.

Author

Delta Wings; Coefficients; Aerodynamic Characteristics; Specimens

19980230627 NASA, Washington, DC USA

A Vibration Absorber for Two-Bladed Helicopters *Un Etouffeur de vibrations pour helicoptere bipale*

Laufer, Th.; Technique et Science Aeronautiques; Nov. 1960, No. 4, pp. 231-235; In English

Report No.(s): NASA-TT-F-43; L-1275; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The report covers the study of and possible use of a dynamic damper. Some theoretical results and a method for calculating the horizontal damper are presented.

Author

Vibration Damping; Research; Numerical Analysis; Procedures

19980231015 Aix-Marseille Univ., Inst. of Fluid Mechanics, Marseille, France

Response of Rotor Lift to an Increase in Collective Pitch in the Case of Descending Flight, the Regime of the Rotor Being Near Autorotation *Reponse de la portance d'un rotor a une augmentation du pas general dans le cas du vol de descente, le regime du rotor etant voisin de l'autorotation*

Valensi, Jacques, Aix-Marseille Univ., France; Rebont, Jean; Soulez-Lariviere, Jean; Comptes Rendus; Apr. 1960; Volume 247, No. 9, pp. 738-741; In English

Report No.(s): NASA-TT-F-18; L-455; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

An elementary calculation inspired by the classic treatment for the steady state permits the determination of the induced velocity and the overall lift of the rotor as a function of the collective pitch for all values of the advance per turn. The nature of the lift response is shown to be essentially a function of the rate of pitch change.

Author

Rotors; Velocity; Rotor Lift; Rotor Aerodynamics; Pitch (Inclination)

19980231066 NASA Langley Research Center, Hampton, VA USA

The Effect of Lift-Drag Ratio and Speed on the Ability to Position a Gliding Aircraft for a Landing on a 5,000-Foot Runway

Reeder, John P., NASA Langley Research Center, USA; Apr. 1959; 12p; In English

Report No.(s): NASA-MEMO-3-12-59L; L-406; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Flight tests were made to determine the capability of positioning a gliding airplane for a landing on a 5,000-foot runway with special reference to the gliding flight of a satellite vehicle of fixed configuration upon reentry into the earth's atmosphere. The

lift-drag ratio and speed of the airplane in the glides were varied through as large a range as possible. The results showed a marked tendency to undershoot the runway when the lift-drag ratios were below certain values, depending upon the speed in the glide. A straight line dividing the successful approaches from the undershoots could be drawn through a lift-drag ratio of about 3 at 100 knots and through a lift-drag ratio of about 7 at 185 knots. Provision of a drag device would be very beneficial, particularly in reducing the tendency toward undershooting at the higher speeds.

Author

Lift Drag Ratio; Velocity; Flight Tests; Positioning; Landing

06

AIRCRAFT INSTRUMENTATION

Includes cockpit and cabin display devices; and flight instruments.

19980228117 NASA Langley Research Center, Hampton, VA USA

Measurement of the Errors of Service Altimeter Installations During Landing-Approach and Take-Off Operations

Gracey, William, NASA Langley Research Center, USA; Jewel, Joseph W., Jr., NASA Langley Research Center, USA; Carpenter, Gene T., NASA Langley Research Center, USA; Nov. 1960; 20p; In English

Report No.(s): NASA-TN-D-463; L-1062; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The overall errors of the service altimeter installations of a variety of civil transport, military, and general-aviation airplanes have been experimentally determined during normal landing-approach and take-off operations. The average height above the runway at which the data were obtained was about 280 feet for the landings and about 440 feet for the take-offs. An analysis of the data obtained from 196 airplanes during 415 landing approaches and from 70 airplanes during 152 take-offs showed that: 1. The overall error of the altimeter installations in the landing- approach condition had a probable value (50 percent probability) of +/- 36 feet and a maximum probable value (99.7 percent probability) of +/- 159 feet with a bias of +10 feet. 2. The overall error in the take-off condition had a probable value of +/- 47 feet and a maximum probable value of +/- 207 feet with a bias of -33 feet. 3. The overall errors of the military airplanes were generally larger than those of the civil transports in both the landing-approach and take-off conditions. In the landing-approach condition the probable error and the maximum probable error of the military airplanes were +/- 43 and +/- 189 feet, respectively, with a bias of +15 feet, whereas those for the civil transports were +/- 22 and +/- 96 feet, respectively, with a bias of +1 foot. 4. The bias values of the error distributions (+10 feet for the landings and -33 feet for the take-offs) appear to represent a measure of the hysteresis characteristics (after effect and recovery) and friction of the instrument and the pressure lag of the tubing-instrument system.

Author

Altimeters; Landing; Errors; Runways; Takeoff

19980228374 NASA Langley Research Center, Hampton, VA USA

Investigation of the Characteristics of an Acceleration-Type Take-Off Indicator in a Large Jet Airplane

Kolnick, Joseph J., NASA Langley Research Center, USA; Rind, Emanuel, NASA Langley Research Center, USA; May 1959; 22p; In English

Report No.(s): NASA-MEMO-4-21-59L; L-173; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The characteristics of a proposed acceleration-type take-off indicator were observed during take-off runs of a large jet airplane. The instrument performed its function satisfactorily. It showed an essentially constant reading, which agreed closely with the predicted value, throughout the take-off except for about the first 135 feet of the ground roll during which the starting windup of the indicator pointer occurred. Although oscillating longitudinal accelerations at the instrument location were as much as +/- 50 percent of the steady-state acceleration, the instrument showed only small excursions from the mean reading equivalent to not more than +/- 5 percent of the mean reading and was considered to be satisfactorily readable.

Author

Experimentation; Acceleration; Takeoff; Dials

07
AIRCRAFT PROPULSION AND POWER

Includes prime propulsion systems and systems components, e.g., gas turbine engines and compressors; and onboard auxiliary power plants for aircraft.

19980227839 NASA Lewis Research Center, Cleveland, OH USA

Experimental Results and Data Format of Preliminary Fan Flutter Investigation Using YF100 Engine

Mehalic, Charles M., NASA Lewis Research Center, USA; Hurrell, Herbert G., NASA Lewis Research Center, USA; Dicus, John H., NASA Lewis Research Center, USA; Lubomski, Joseph F., NASA Lewis Research Center, USA; Kurkov, Anatole P., NASA Lewis Research Center, USA; Evans, David G., NASA Lewis Research Center, USA; Apr. 1977; 102p; In English
Report No.(s): NASA-TM-SX-3444; E-8877; No Copyright; Avail: CASI; A06, Hardcopy; A02, Microfiche

A preliminary investigation was conducted to determine the conditions which can cause flutter to occur in the first-stage rotor of the fan of a turbofan engine. Strain gages and stagewise aerodynamic instrumentation were installed in the fan of a YF100 engine. The engine was operated over the low corrected speed range of the fan map, from below its normal operating line to near stall, and over a range of inlet guide vane angles, pressures, and temperatures. Flutter was encountered, and the characteristics of the six flutter points and the associated aerodynamic conditions were recorded. Comparisons with the 97 nonflutter points recorded were made. The data format is presented to assist in future data transmittals and analysis. A companion report, NASA TM X-3508, describes the engine modifications, strain-gage instrumentation, and data acquisition system used in the investigation.

Author

Data Acquisition; Strain Gages; Turbofan Engines; Flutter; Format

19980227877 NASA Langley Research Center, Hampton, VA USA

Static Thrust of an Annular Nozzle with a Concave Central Base

Corson, Blake W., Jr., NASA Langley Research Center, USA; Mercer, Charles E., NASA Langley Research Center, USA; Sep. 1960; 20p; In English
Report No.(s): NASA-TN-D-418; L-851; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A static test of an annular nozzle with a concave central base, producing a jet in which tangents to the jet streamlines at the exit converged toward a region on the axis of symmetry downstream of the exit, has indicated good thrust performance. A value of nozzle-flow coefficient only slightly less than unity indicates the internal loss to be small. Pressures on the concave central base are relatively large and positive, and a predictable portion of the total thrust of the jet is exerted on the central base.

Author

Annular Nozzles; Static Thrust; Static Tests; Nozzle Flow

19980227966 NASA Lewis Research Center, Cleveland, OH USA

Experimental Investigation of a 0.35 Hub-Tip Radius Ratio Transonic Axial Flow Rotor Designed for 40 Pounds per Second per Square Foot with a Design Tip Diffusion Factor of 0.20

Yasaki, Paul T., NASA Lewis Research Center, USA; Montgomery, John C., NASA Lewis Research Center, USA; Sep. 1959; 34p; In English
Report No.(s): NASA-TM-X-86; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

In order to determine the effect of a low design diffusion factor on the performance of a transonic axial-flow compressor rotor, a high-specific-flow rotor with a 0.35 hub-tip radius ratio was designed, fabricated and tested. This rotor used a design tip diffusion factor of 0.20 with a design corrected specific weight flow of 40 pounds per second per square foot of frontal area, a total-pressure ratio of 1.27, and an adiabatic efficiency of 0.96. The design, rotor performance, and blade element performance are presented with a discussion on rotor shock losses and a comparison with a similarly designed rotor with a tip diffusion factor of 0.35. At the design corrected tip speed of 1100 feet per second, a peak rotor adiabatic efficiency of 0.88 was attained at a corrected specific weight flow of 39 pounds per second per square foot of frontal area with a mass-averaged total-pressure ratio of 1.27. The blade element tip diffusion factor was 0.281, which is 0.08 higher than the design value of 0.20. Peak efficiencies of 0.95, 0.91, 0.89, and 0.85 were obtained at 70, 80, 90, and 110 percent of design speed, respectively. Comparison of the performance of the rotor reported herein and a similarly designed rotor with increased blade loading indicates that higher blade loading results in a more desirable rotor because of a higher pressure ratio and equivalent efficiency. Computed values of shock losses at the rotor tip section

indicate that the losses at peak efficiency are primarily a function of shock losses since the profile losses are only a small percentage of the total loss.

Author

Compressor Rotors; Axial Flow; Transonic Flow; Pressure Ratio; Blade Tips; Turbocompressors; Transonic Compressors

19980228162 NASA Dryden Flight Research Center, Edwards, CA USA

Initial Flight Test Evaluation of the F-15 ACTIVE Axisymmetric Vectoring Nozzle Performance

Orme, John S., NASA Dryden Flight Research Center, USA; Hathaway, Ross, Analytical Services and Materials, Inc., USA; Ferguson, Michael D., Pratt and Whitney Aircraft, USA; Jul. 1998; 22p; In English; 34th; Propulsion, 13-15 Jul. 1998, Cleveland, OH, USA; Sponsored by American Inst. of Aeronautics and Astronautics, USA

Contract(s)/Grant(s): RTOP 529-50-40-00-32

Report No.(s): NASA/TM-1998-206558; H-2267; NAS 1.15:206558; AIAA Paper 98-3871; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A full envelope database of a thrust-vectoring axisymmetric nozzle performance for the Pratt & Whitney Pitch/Yaw Balance Beam Nozzle (P/YBBN) is being developed using the F-15 Advanced Control Technology for Integrated Vehicles (ACTIVE) aircraft. At this time, flight research has been completed for steady-state pitch vector angles up to 20° at an altitude of 30,000 ft from low power settings to maximum afterburner power. The nozzle performance database includes vector forces, internal nozzle pressures, and temperatures all of which can be used for regression analysis modeling. The database was used to substantiate a set of nozzle performance data from wind tunnel testing and computational fluid dynamic analyses. Findings from initial flight research at Mach 0.9 and 1.2 are presented in this paper. The results show that vector efficiency is strongly influenced by power setting. A significant discrepancy in nozzle performance has been discovered between predicted and measured results during vectoring.

Author

Flight Tests; Nozzle Efficiency; Technology Assessment; Wind Tunnel Tests; Pitch (Inclination); Computational Fluid Dynamics

19980228223 NASA Lewis Research Center, Cleveland, OH USA

Investigation of a 4.5-Inch-Mean-Diameter Two-Stage Axial-Flow Turbine Suitable for Auxiliary Power Drives

Wong, Robert Y., NASA Lewis Research Center, USA; Monroe, Daniel E., NASA Lewis Research Center, USA; Mar. 1959; 28p; In English

Report No.(s): NASA-MEMO-4-6-59E; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The design and experimental investigation of a 4.5-inch-mean-diameter two-stage turbine are presented herein and used to study the effect of size on the efficiency of turbines in the auxiliary power drive class. The results of the experimental investigation indicated that design specific work was obtained at design speed at a total-to-static efficiency of 0.639. At design pressure ratio, design static-pressure distribution through the turbine was obtained with an equivalent specific work output of 33.2 Btu per pound and an efficiency of 0.656. It was found that, in the design of turbines in the auxiliary power drive class, Reynolds number plays an important part in the selection of the design efficiency. Comparison with theoretical efficiencies based on a loss coefficient and velocity diagrams are presented. Close agreement was obtained between theory and experiment when the loss coefficient was adjusted for changes in Reynolds number to the $-1/5$ power.

Author

Experimentation; Two Stage Turbines; Design Analysis; Efficiency

19980228229 NASA Lewis Research Center, Cleveland, OH USA

Performance of Typical Rear-Stage Axial-Flow Compressor Rotor Blade Row at Three Different Blade Setting Angles

Kussoy, Marvin I., NASA Lewis Research Center, USA; Bachkin, Daniel, NASA Lewis Research Center, USA; Jan. 1959; 44p; In English

Report No.(s): NASA-MEMO-11-27-58E; E-117; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A comparison of the performance of a single-stage rotor run at three different blade setting angles is presented. The rotor was of a design typical for a last stage of a multistage compressor. At each setting angle, the rotor blade row was operated from 53 to 100 percent of equivalent maximum speed (850 ft/sec tip speed) at constant inlet pressure. Hot-wire anemometry was used to observe rotating-stall and surge patterns in time unsteady flow. Results indicated that an increase in peak pressure ratio and an increase in maximum equivalent weight flow were obtained at each speed investigated when the blade setting angle was decreased. An increase in peak efficiency was achieved with decrease in blade setting angle for part of the range of speeds investigated. However, the peak efficiencies for the three blade setting angles were approximately the same at the maximum speed investigated. The flow ranges for all three configurations were about the same at minimum speed and decreased at almost the same rate when the

rotative speed was increased through part of the range of speeds investigated. At maximum speed, the flow range for the smallest setting angle was considerably less than the flow range for the other two configurations. A decrease in efficiency and flow range for the smallest blade setting angle at maximum speed can be attributed primarily to a Mach number effect. In addition, because of the difference in projected axial chord lengths at the casing wall, some effect on performance could be expected from the change in three-dimensional flow occurring at the tip. Rotating-stall characteristics for the two smaller blade setting angles were essentially the same. Only surge could be detected for the largest blade setting angle in the unstable-flow region of operation.

Author

Compressor Rotors; Engine Parts; Turbocompressors; Engine Design; Three Dimensional Flow; Angles (Geometry); Compressor Efficiency

19980228230 NASA Lewis Research Center, Cleveland, OH USA

Comparison of Calculated and Experimental Total-Pressure Loss and Airflow Distribution in Tubular Turbojet Combustors with Tapered Liners

Grobman, Jack S., NASA Lewis Research Center, USA; Jan. 1959; 50p; In English

Report No.(s): NASA-MEMO-11-26-58E; E-126; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Incompressible-flow calculations were performed to determine the effects of combustor geometric and operating variables on pressure loss and airflow distribution in a tubular combustor with a tapered liner. The calculations include the effects of momentum transfer between annulus and liner gas streams, annulus wall friction, heat release, and discharge coefficients of liner air-entry holes. Generalized curves are presented which show the effects of liner-wall inclination, liner open hole area, and temperature rise across the combustor on pressure loss and airflow distribution for a representative parabolic liner hole distribution. A comparison of the experimental data from 12 tapered liners with the theoretical calculations indicates that reasonable design estimates can be made from the generalized curves. The calculated pressure losses of the tapered liners are compared with those previously reported for tubular liners.

Author

Combustion Chambers; Incompressible Flow; Linings; Flow Distribution; Turbojet Engines; Air Flow; Pressure Reduction

19980228231 NASA Lewis Research Center, Cleveland, OH USA

Engine Operating Conditions that Cause Thermal-Fatigue Cracks in Turbojet-Engine Buckets

Johnston, James R., NASA Lewis Research Center, USA; Weeton, John W., NASA Lewis Research Center, USA; Signorelli, Robert A., NASA Lewis Research Center, USA; Apr. 1959; 28p; In English

Report No.(s): NASA-MEMO-4-7-59E; E-281; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Five engine tests were conducted to definitely establish the failure mechanism of leading-edge cracking and to determine which conditions of engine operation cause the failures. Five groups of S-616 and M-252 buckets from master lots were run consecutively in the same J47-25 engine. The tests included a steady-state run at full-power conditions, rapid cycling between idle and rated speed, and three different start-stop tests. The first start-stop test consisted of cycles of start and stop with 5 minutes of idle speed before each stop; the second included cycles of start and stop but with 15 minutes of rated speed before each stop; the third consisted of cycles of gradual starts and normal stops with 5 minutes at idle speed before each stop. The test results demonstrated that the primary cause of leading-edge cracking was thermal fatigue produced by repeated engine starts. The leading edge of the bucket experiences plastic flow in compression during starts and consequently is subjected to a tensile stress when the remainder of the bucket becomes heated and expands. Crack initiation was accelerated when rated-speed operation was added to each normal start-stop cycle. This acceleration of crack formation was attributed to localized creep damage and perhaps to embrittlement resulting from overaging. It was demonstrated that leading-edge cracking can be prevented simply by starting the engine gradually.

Author

Engine Tests; Thermal Fatigue; Turbojet Engines; Engine Failure; Leading Edges; Crack Initiation; Turbine Blades

19980228284 NASA Lewis Research Center, Cleveland, OH USA

Experimental Investigation of an Air-Cooled Turbine Operating in a Turbojet Engine at Turbine Inlet Temperatures up to 2500 F

Cochran, Reeves P., NASA Lewis Research Center, USA; Dengler, Robert P., NASA Lewis Research Center, USA; Jul. 1961; 48p; In English

Report No.(s): NASA-TN-D-1046; E-1163; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An experimental investigation was made of an air-cooled turbine at average turbine inlet temperatures up to 2500 F. A modified production-model 12-stage axial-flow-compressor turbojet engine operating in a static sea-level stand was used as the test

vehicle. The modifications to the engine consisted of the substitution of special combustor and turbine assemblies and double-walled exhaust ducting for the standard parts of the engine. All of these special parts were air-cooled to withstand the high operating temperatures of the investigation. The air-cooled turbine stator and rotor blades were of the corrugated-insert type. Leading-edge tip caps were installed on the rotor blades to improve leading-edge cooling by diverting the discharge of coolant to regions of lower gas pressure toward the trailing edge of the blade tip. Caps varying in length from 0.15- to 0.55-chord length were used in an attempt to determine the optimum cap length for this blade. The engine was operated over a range of average turbine inlet temperatures from about 1600 to about 2500 F, and a range of average coolant-flow ratios of 0.012 to 0.065. Temperatures of the air-cooled turbine rotor blades were measured at all test conditions by the use of thermocouples and temperature-indicating paints. The results of the investigation indicated that this type of blade is feasible for operation in turbojet engines at the average turbine inlet temperatures and stress levels tested (maximums of 2500 F and 24,000 psi, respectively). An average one-third-span blade temperature of 1300 F could be maintained on 0.35-chord tip cap blades with an average coolant-flow ratio of about 0.022 when the average turbine inlet temperature was 2500 F and cooling-air temperature was about 260 F. All of the leading-edge tip cap lengths improved the cooling of the leading-edge region of the blades, particularly at low average coolant-flow ratios. At high gas temperatures, such parts as the turbine stator and the combustor liners are likely to be as critical as the turbine rotor blades.

Author

Air Cooling; Turbojet Engines; Inlet Temperature; Trailing Edges; Thermocouples

19980228313 NASA Lewis Research Center, Cleveland, OH USA

Exploratory Investigation of Aerodynamic Flameholders for Afterburner Application

Butze, Helmut F., NASA Lewis Research Center, USA; Metzler, Allen J., NASA Lewis Research Center, USA; May 1959; 12p; In English

Report No.(s): NASA-MEMO-4-9-59E; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation was conducted to determine the flameholding capabilities of aerodynamic jets at afterburner operating conditions. Stability data for a number of aerodynamic flameholders were obtained in a 5- by 5-inch test section at inlet-air reference velocities up to 600 feet per second, an inlet-air temperature of 1250 F, and a combustor-inlet pressure of 15 inches of mercury absolute. Combustion efficiency and stability data of the more promising combinations were then obtained in a 10- by 12-inch test section at the same test conditions. Both air and stoichiometric mixtures of fuel and air were used in the jets; mixture flow rates were approximately 1 percent by weight of the total air-flow rate. Injection pressures were limited to values that might be available from compressor-bleed air. At a reference velocity of 600 feet per second, aerodynamic flame-holders alone were unable to maintain a stable flame at injection pressures up to 70 pounds per square inches large reductions in velocity were required to achieve flame stabilization. When the aerodynamic jets were used in combination with a V-gutter flameholder with approximately a 30 percent blocked area, flame stabilization was attained at a velocity of 600 feet per second; however, the combustion efficiencies of the various combinations were no greater than that obtained with the V-gutter alone.

Author

Flame Holders; Afterburning; Flame Stability; Combustion Chambers; Air Flow; Stabilization

19980228322 NASA Lewis Research Center, Cleveland, OH USA

Air-Cooled Turbine Blades with Tip Cap For Improved Leading-Edge Cooling

Calvert, Howard F., NASA Lewis Research Center, USA; Meyer, Andre J., Jr., NASA Lewis Research Center, USA; Morgan, William C., NASA Lewis Research Center, USA; Feb. 1959; 28p; In English

Report No.(s): NASA-MEMO-2-9-59E; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation was conducted in a modified turbojet engine to determine the cooling characteristics of the semistrut corrugated air-cooled turbine blade and to compare and evaluate a leading-edge tip cap as a means for improving the leading-edge cooling characteristics of cooled turbine blades. Temperature data were obtained from uncapped air-cooled blades (blade A), cooled blades with the leading-edge tip area capped (blade B), and blades with slanted corrugations in addition to leading-edge tip caps (blade C). All data are for rated engine speed and turbine-inlet temperature (1660 F). A comparison of temperature data from blades A and B showed a leading-edge temperature reduction of about 130 F that could be attributed to the use of tip caps. Even better leading-edge cooling was obtained with blade C. Blade C also operated with the smallest chordwise temperature gradients of the blades tested, but tip-capped blade B operated with the lowest average chordwise temperature. According to a correlation of the experimental data, all three blade types could operate satisfactorily with a turbine-inlet temperature of 2000 F and a coolant flow of 3 percent of engine mass flow or less, with an average chordwise temperature limit of 1400 F. Within the range

of coolant flows investigated, however, only blade C could maintain a leading-edge temperature of 1400 F for a turbine-inlet temperature of 2000 F.

Author

Air Cooling; Turbine Blades; Leading Edges; Engine Inlets; Inlet Temperature; Coolants; Cooling Systems

19980228408 NASA Lewis Research Center, Cleveland, OH USA

Experimental Evaluation of Cermet Turbine Stator Blades for Use at Elevated Gas Temperatures

Chiarito, Patrick T., NASA Lewis Research Center, USA; Johnston, James R., NASA Lewis Research Center, USA; Feb. 1959; 28p; In English

Report No.(s): NASA-MEMO-2-13-59E; E-147; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The suitability of cermets for turbine stator blades of a modified turbojet engine was determined at an average turbine-inlet-gas temperature of 2000 F. Such an increase in temperature would yield a premium in thrust from a service engine. Because the cermet blades require no cooling, all the available compressor bleed air could be used to cool a turbine made from conventional ductile alloys. Cermet blades were first run in 100-hour endurance tests at normal gas temperatures in order to evaluate two methods for mounting them. The elevated gas-temperature test was then run using the method of support considered best for high-temperature operation. After 52 hours at 2000 F, one of the group of four cermet blades fractured probably because of end loads resulting from thermal distortion of the spacer band of the nozzle diaphragm. Improved design of a service engine would preclude this cause of premature failure.

Author

Evaluation; Experimentation; Turbine Blades; Stator Blades; Cermets; Failure

19980228409 NASA Lewis Research Center, Cleveland, OH USA

Effect of Stator and Rotor Aspect Ratio on Transonic-Turbine Performance

Wong, Robert Y., NASA Lewis Research Center, USA; Monroe, Daniel E., NASA Lewis Research Center, USA; Feb. 1959; 30p; In English

Report No.(s): NASA-MEMO-2-11-59E; E-177; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The effect of stator and rotor aspect ratio on transonic-turbine performance was experimentally investigated. The stator aspect ratios covered were 1.6, 0.8, and 0.4, while the rotor aspect ratios investigated were 1.46 and 0.73. It was found that the observed variation in turbine design-point efficiency was negligible. Thus, within the range of aspect ratio investigated, these results verify for turbines operating in the transonic flow range the finding of a reference report, which showed analytically that, if blade shape and solidity are held constant, the aspect ratio may be varied over a wide range without appreciable change in turbine efficiency.

Author

Aspect Ratio; Rotors; Stators; Experimentation

08

AIRCRAFT STABILITY AND CONTROL

Includes aircraft handling qualities; piloting; flight controls; and autopilots.

19980227822 NASA Langley Research Center, Hampton, VA USA

Supersonic Jet Tests of Missile Stabilizers

Vosteen, Louis F., NASA Langley Research Center, USA; Rosecrans, Richard, NASA Langley Research Center, USA; Dec. 1959; 28p; In English

Report No.(s): NASA-TM-X-121; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Seven stabilizers were tested at a Mach number of 2 in order to determine the effects of aerodynamic heating and loading on the structural stability of the stabilizer. The models differed in internal structure and postcure temperatures of the laminated Fiberglass skin. Tests were made at various stagnation temperatures between 440 F and 625 F. The postcure temperatures of the Fiberglass skins were found to affect significantly the ability of the model to withstand the imposed test conditions.

Author

Aerodynamic Heating; Horizontal Tail Surfaces; Structural Analysis; Supersonic Speed; Temperature Effects; Supersonic Jet Flow; Wind Tunnel Tests

19980227823 NASA Langley Research Center, Hampton, VA USA

Stability Investigation of a Blunted Cone and a Blunted Ogive with a Flared Cylinder Afterbody at Mach Numbers from 0.30 to 2.85

Coltrane, Lucille C., NASA Langley Research Center, USA; Nov. 1959; 30p; In English

Report No.(s): NASA-TM-X-199; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A cone with a blunt nose tip and a 10.7 deg cone half angle and an ogive with a blunt nose tip and a 20 deg flared cylinder afterbody have been tested in free flight over a Mach number range of 0.30 to 2.85 and a Reynolds number range of $1 \times 10^{(exp 6)}$ to $23 \times 10^{(exp 6)}$. Time histories, cross plots of force and moment coefficients, and plots of the longitudinal force, coefficient, rolling velocity, aerodynamic center, normal- force-curve slope, and dynamic stability are presented. With the center-of-gravity location at about 50 percent of the model length, the models were both statically and dynamically stable throughout the Mach number range. For the cone, the average aerodynamic center moved slightly forward with decreasing speeds and the normal-force-curve slope was fairly constant throughout the speed range. For the ogive, the average aerodynamic center remained practically constant and the normal-force-curve slope remained practically constant to a Mach number of approximately 1.6 where a rising trend is noted. Maximum drag coefficient for the cone, with reference to the base area, was approximately 0.6, and for the ogive, with reference to the area of the cylindrical portion, was approximately 2.1.

Author

Ogives; Cylindrical Bodies; Stability; Moment Distribution; Aerodynamic Drag; Afterbodies; Aerodynamic Coefficients; Aerodynamic Balance

19980227828 NASA Langley Research Center, Hampton, VA USA

Static Longitudinal Stability of a Rocket Vehicle Having a Rear-Facing Step Ahead of the Stabilizing Fins

Keynton, Robert J., NASA Langley Research Center, USA; Nov. 1961; 24p; In English

Report No.(s): NASA-TN-D-993; L-1836; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Tests were conducted at Mach numbers of 3.96 and 4.65 in the Langley Unitary Plan wind tunnel to determine the static longitudinal stability characteristics of a fin-stabilized rocket-vehicle configuration which had a rearward facing step located upstream of the fins. Two fin sizes and planforms, a delta and a clipped delta, were tested. The angle of attack was varied from 6 deg to -6 deg and the Reynolds number based on model length was about 10×10^6 . The configuration with the larger fins (clipped delta) had a center of pressure slightly rearward of and an initial normal-force-curve slope slightly higher than that of the configuration with the smaller fins (delta) as would be expected. Calculations of the stability parameters gave a slightly lower initial slope of the normal-force curve than measured data, probably because of boundary-layer separation ahead of the step. The calculated center of pressure agreed well with the measured data. Measured and calculated increments in the initial slope of the normal-force curve and in the center of pressure, due to changing fins, were in excellent agreement indicating that separated flow downstream of the step did not influence flow over the fins. This result was consistent with data from schlieren photographs.

Author

Static Stability; Longitudinal Stability; Boundary Layer Separation; Stabilization; Separated Flow; Rocket Vehicles

19980227835 NASA Dryden Flight Research Center, Edwards, CA USA

Analysis of a Pilot Airplane Lateral Instability Experienced with the X-15 Airplane

Taylor, Lawrence W., Jr., NASA Dryden Flight Research Center, USA; Nov. 1961; 32p; In English

Report No.(s): NASA-TN-D-1059; H-225; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An analysis is made of a lateral-control problem in which the pilot, through normal application of control, induces divergent oscillations in bank angle. The problem, first encountered on the X-15 simulator and later confirmed in flight, is explained through the use of root-locus plots of the pilot-airplane combination in which the pilot is represented by a human transfer function. A parameter is developed which is useful for predicting the lateral-control problem and for showing the effect of the principal aerodynamic and inertial parameters. Also, means of determining regions in the flight envelope where the pilot-airplane would be susceptible to lateral instability are developed.

Author

X-15 Aircraft; Lateral Control; Pilot Induced Oscillation; Longitudinal Control; Control Stability; Aircraft Control

19980227858 NASA Langley Research Center, Hampton, VA USA

Analytical Investigation of an Adaptive Flight-Control System Using a Sinusoidal Test Signal

Harris, Jack E., NASA Langley Research Center, USA; Jun. 1961; 64p; In English

Report No.(s): NASA-TN-D-909; L-1456; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An analytical study was made of an adaptive flight-control system which measures vehicle response to small-amplitude control-surface deflections produced by a sinusoidal test signal. Changes in the response to this signal are related to environmental changes, and the system is continuously altered to maintain this response equal to a preselected value. The system is suitable for use in high-performance aircraft and missiles and requires only the addition of a signal generator and a logic circuit consisting of a filter-rectifier network and a comparator-integrator network to a basic command-control system. Thus, it presents a relatively simple approach to the problem. The effects on system performance of variation in flight condition, system-gain level, test-signal frequency, and sensor location are included in the analysis. Longitudinal control of a high-performance research aircraft over flight conditions ranging from landing approach to a Mach number of 5.8 at an altitude of 150,000 feet, and longitudinal control of a four-stage solid-fuel missile including the first bending mode over the atmospheric portion of a launch trajectory constituted the basis for the analytical study. Results of an analog-computer study using time-varying coefficients are presented to compare the control obtained with the adaptive system with that obtained with a fixed-gain system during the atmospheric portion of a missile launch trajectory. The system has demonstrated an ability to maintain satisfactory vehicle control-system stability over wide ranges of environmental change.

Author

Adaptive Control; Sine Waves; Longitudinal Control; Flight Control; Signal Generators

19980227861 NASA Langley Research Center, Hampton, VA USA

Calculated Responses of a Large Sweptwing Airplane to Continuous Turbulence with Flight-Test Comparisons

Bennett, Floyd V., NASA Langley Research Center, USA; Pratt, Kermit G., NASA Langley Research Center, USA; 1960; 28p; In English

Report No.(s): NASA-TR-R-69; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Calculated responses of symmetrical airplane motions, wing deformations, and wing loads due to gusts are shown to compare favorably with available flight-test results. These calculated responses are based on random-process theory, five degrees of freedom, lifting-surface aerodynamics, and one-dimensional vertical turbulence. The extent to which various degrees of freedom contribute to the responses is examined and in this connection the relative effects of static and dynamic aeroelasticity are determined.

Author

Swept Wings; Aeroelasticity; Aerodynamics; Turbulence; Flight Tests; Degrees of Freedom

19980227871 NASA Langley Research Center, Hampton, VA USA

Static Stability Characteristics of Three Thick Wing Models with Parabolic Plan Forms at a Mach Number of 3.11

Queijo, M. J., NASA Langley Research Center, USA; Oct. 1959; 20p; In English

Report No.(s): NASA-TM-X-141; L-597; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An experimental investigation has been made to determine the static stability characteristics of three thick wing models with parabolic plan forms at a Mach number of 3.11 for angles of attack from about -6 to 16 deg. The primary variable was aspect ratio, with the plan-form area and the ratio of base height to span kept the same for all three models. All models had stable, linear pitching-moment curves about the quarter chord of the wing mean aerodynamic chord. The model with the lowest aspect ratio attained a maximum untrimmed lift-drag ratio of about 5.0 at an angle of attack of about 8 deg. Increasing the aspect ratio (which was accompanied by an increase in base area because the ratio of the base height to span was kept constant) caused a decrease in maximum lift-drag ratio. All models were directionally stable for the range of angle of attack of the tests. Addition of a vertical tail to the models caused an increase in the directional stability over the angle-of-attack range. In general, the lateral aerodynamic characteristics of the models were not linear functions of angle of attack over any appreciable angle-of-attack range.

Author

Static Stability; Supersonic Speed; Aerodynamic Characteristics; Directional Stability; Wings; Gliders; Wind Tunnel Models; Wind Tunnel Tests

19980227872 NASA Langley Research Center, Hampton, VA USA

Low-Speed Measurements of Oscillatory Lateral Stability Derivatives of a 1/7-Scale Model of the North American X-15 Airplane

Paulson, John W., NASA Langley Research Center, USA; Hassell, James L., Jr., NASA Langley Research Center, USA; Nov. 1959; 30p; In English

Report No.(s): NASA-TM-X-144; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation to determine the low-speed rolling, yawing, and sideslipping derivatives of a 1/7-scale model which was used to represent the original configuration and a modified configuration of the North American X-15 airplane has been conducted in the Langley free-flight tunnel. The original model was modified to approximately represent the final airplane configuration

by reducing the size of the fuselage side fairings and changing the vertical-tail arrangement. The effects of various tail arrangements were determined for both configurations and the effect of small forebody strakes was determined for the modified configuration only.

Author

Roll; Yaw; Sideslip; Lateral Stability; Aerodynamic Stability; Scale Models; Wind Tunnel Tests; Stability Derivatives; Free Flight; Forebodies; Aerodynamic Configurations; Aircraft Structures

19980227875 NASA Langley Research Center, Hampton, VA USA

Static Stability Characteristics of a Series of Hypersonic Boost-Glide Configurations at Mach Numbers of 1.41 and 2.01

Foster, Gerald V., NASA Langley Research Center, USA; Nov. 1959; 88p; In English

Report No.(s): NASA-TM-X-167; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

An investigation of the static stability characteristics of several hypersonic boost-glide configurations has been conducted in the Langley 4- by 4-foot supersonic pressure tunnel at Mach numbers of 1.41 and 2.01 (with Reynolds numbers per foot of $2.90 \times 10(\text{exp } 6)$ and $2.41 \times 10(\text{exp } 6)$ respectively). This series of configurations consisted of a cone, with and without cruciform fins, a trihedron, two low-aspect-ratio delta wings that differed primarily in cross-sectional shape, and two wing-body configurations. All configurations indicated reasonably linear pitching-, yawing-, and rolling-moment characteristics for angles of attack to at least 12 deg. The maximum lift-drag ratio for the zero-thrust condition (base drag included) was about 3 for the delta-wing configurations and about 4 for the wing-body configurations.

Author

Boostglide Vehicles; Hypersonic Vehicles; Static Stability; Wind Tunnel Tests; Body-Wing Configurations; Supersonic Speed

19980227971 NASA Langley Research Center, Hampton, VA USA

Flight Investigation of a Normal-Acceleration Automatic Longitudinal Control System in a Fighter Airplane

Sjoberg, S. A., NASA Langley Research Center, USA; Russell, Walter R., NASA Langley Research Center, USA; Alford, William L., NASA Langley Research Center, USA; Dec. 1958; 44p; In English

Report No.(s): NASA-MEMO-1-10-26-58L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A flight investigation was made to obtain experimental information on the handling qualities of a normal-acceleration type of automatic longitudinal control system. The control system was installed in a subsonic fighter-type airplane. In hands-off (stick-free) flight the normal-acceleration control system attempted to regulate the normal acceleration to a constant value which is dependent on the automatic-control-system trim setting. In maneuvering flight a given pilot's stick deflection produced a proportional change in normal acceleration, the change in acceleration being independent of flight condition. A small side-located controller stick was used by the pilot to introduce signals into the automatic control system. In the flight program emphasis was placed on the acceleration-limiting capabilities of the control system. The handling qualities were investigated in maneuvers such as slow and rapid pull-ups and turns and also in flight operations such as cruising, stalls, landings, aerobatics, and air-to-air tracking. Good acceleration limiting was obtained with the normal-acceleration control system by limiting the magnitude of the input signal that the pilot could introduce into the control system. The same values of control-system gain settings could be used from an acceleration-limiting stand-point at both 10,000 and 30,000 feet for the complete speed range of the airplane. The response characteristics of the airplane-control system combination were also satisfactory at both high and low altitude with these same values of control-system gain setting. In the pilot's opinion, the normal-acceleration control system provided good stability and control characteristics in flight operations such as cruising, stalls, landings, aerobatics, and air-to-air tracking.

Author

Flight Tests; Longitudinal Control; Automatic Control; Fighter Aircraft; Flight Control; Free Flight; Flight Operations; Controllers; Aircraft Maneuvers; Longitudinal Stability

19980227983 NASA Langley Research Center, Hampton, VA USA

Subsonic Kernel-Function Flutter Analysis of a Highly Tapered Tail Surface and Comparison with Experimental Results

Walberg, Gerald D., NASA Langley Research Center, USA; Sep. 1960; 38p; In English

Report No.(s): NASA-TN-D-379; L-615; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A flutter analysis employing the kernel function for three-dimensional, subsonic, compressible flow is applied to a flutter-tested tail surface which has an aspect ratio of 3.5, a taper ratio of 0.15, and a leading-edge sweep of 30 deg. Theoretical and experimental results are compared at Mach numbers from 0.75 to 0.98. Good agreement between theoretical and experimental flutter dynamic pressures and frequencies is achieved at Mach numbers to 0.92. At Mach numbers from 0.92 to 0.98, however, a second

solution to the flutter determinant results in a spurious theoretical flutter boundary which is at a much lower dynamic pressure and at a much higher frequency than the experimental boundary.

Author

Flutter Analysis; Kernel Functions; Flutter; Tail Surfaces; Leading Edge Sweep; Compressible Flow; Functional Analysis; Subsonic Flow; Tapering; Three Dimensional Flow

19980227989 NASA Langley Research Center, Hampton, VA USA

Determination of Lateral Stability Characteristics from Free-Flight Model Tests, with Experimental Results on the Effects of Wing Vertical Position and Dihedral at Transonic Speeds

Gillis, Clarence L., NASA Langley Research Center, USA; Mitchell, Jesse L., NASA Langley Research Center, USA; DAiutolo, Charles T., NASA Langley Research Center, USA; 1960; 40p; In English

Report No.(s): NASA-TR-R-65; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A test and analysis method is presented for determining airplane lateral stability characteristics, including aerodynamic derivatives, from flight tests of scale models. The method of analysis utilizes the rotating time-vector concept and also a quasi-static approach. Data are presented at transonic speeds for three swept-wing rocket-propelled models differing only in vertical position and dihedral of the wing. The method proved to be adequate for delineating the major effects of the geometric variations on the aerodynamic lateral stability derivatives. The effects of Reynolds number on the linearity of the static stability data for an unswept-wing configuration are illustrated.

Author

Aerodynamic Configurations; Aerodynamic Stability; Transonic Speed; Swept Wings; Static Stability; Scale Models

19980227996 NASA Langley Research Center, Hampton, VA USA

Effects of Control-Feel Configuration on Airplane Longitudinal Control Response

Crane, Harold L., NASA Langley Research Center, USA; Sommer, Robert W., NASA Langley Research Center, USA; Oct. 1961; 38p; In English

Report No.(s): NASA-TN-D-912; L-1641; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A general study of longitudinal control feel was made with a transonic fighter-type airplane equipped with a control-feel system which was adjustable in flight. The control-feel system provided a feel component with individual gain control in proportion to each of five quantities: stick deflection, stick rate, airplane normal acceleration, pitching acceleration, and pitching velocity. A number of feel configurations were investigated in flight and analytically. These feel configurations had feel components in various amounts from various combinations of these five sources. The results contained herein are all for an airplane center-of-gravity position at approximately 25 percent of the mean aerodynamic chord, a Mach number of 0.85, and an altitude of 28,000 feet. Results are presented as time histories, as plots of the variation of peak force per g with input duration, and as frequency-response plots. A number of frequency-response plots are included to illustrate the effects of choice of feel sources and gains. The results illustrate the desirability of balancing a normal-acceleration feel component with a pitching-acceleration feel component. Pitching-velocity feel is shown to be useful for shaping control-system frequency response. The results suggest the desirability of designing a control-feel system to a large extent by means of frequency-response analysis in order to keep the shapes of the frequency-response curves within desirable limits.

Author

Longitudinal Control; Fighter Aircraft; Airfoil Profiles; Center of Gravity; Chords (Geometry)

19980227997 NASA Langley Research Center, Hampton, VA USA

Investigation of the Low-Subsonic Stability and Control Characteristics of a Free-Flying Model of a Thick 70 deg Delta Reentry Configuration

Paulson, John W., NASA Langley Research Center, USA; Shanks, Robert E., NASA Langley Research Center, USA; Oct. 1961; 36p; In English

Report No.(s): NASA-TN-D-913; L-1684; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation of the low-subsonic flight characteristics of a thick 70 deg delta reentry configuration having a diamond cross section has been made in the Langley full-scale tunnel over an angle-of-attack range from 20 to 45 deg. Flight tests were also made at angles of attack near maximum lift ($\alpha = 40$ deg) with a radio-controlled model dropped from a helicopter. Static and dynamic force tests were made over an angle-of-attack range from 0 to 90 deg. The longitudinal stability and control characteristics were considered satisfactory when the model had positive static longitudinal stability. It was possible to fly the model with a small amount of static instability, but the longitudinal characteristics were considered unsatisfactory in this condition. At angles of attack above the stall the model developed a large, constant-amplitude pitching oscillation. The lateral stability characteristics

were considered to be only fair at angles of attack from about 20 to 35 deg because of a lightly damped Dutch roll oscillation. At higher angles of attack the oscillation was well damped and the lateral stability was generally satisfactory. The Dutch roll damping at the lower angles of attack was increased to satisfactory values by means of a simple rate-type roll damper. The lateral control characteristics were generally satisfactory throughout the angle- of-attack range, but there was some deterioration in aileron effectiveness in the high angle-of-attack range due mainly to a large increase in damping in roll.

Author

Flight Characteristics; Longitudinal Stability; Lateral Stability; Subsonic Speed; Lateral Control; Static Tests; Dynamic Tests

19980227999 NASA Langley Research Center, Hampton, VA USA

Stability and Control Characteristics of a Small-Scale Model of an Aerial Vehicle Supported by Two Ducted Fans

Parlett, Lysle P., NASA Langley Research Center, USA; Jul. 1961; 24p; In English

Report No.(s): NASA-TN-D-920; L-1481; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been made to determine the stability and control characteristics in hovering and in forward flight of a free-flight model representing a type of vertical-take-off-and-landing aircraft which utilizes two fixed ducted fans as its sole source of lift and propulsion. The model, having fans 28 inches in diameter, was considered to be approximately one-third the size of a full-scale aircraft. Control moments for most of the hovering tests and all the forward-flight tests were provided by remotely controlled compressed-air jets at the sides and ends of the model. For one brief phase of the hovering investigation a system of vanes in the duct slipstreams was substituted for the jets as a means of roll control. During the forward-flight tests, the model was flown with both the tandem and side-by-side duct arrangements. In hovering the model exhibited strongly divergent oscillations about the pitch and roll axes. The pitching oscillation of the tandem configuration was of a fairly long period and was not particularly difficult to control; the rolling oscillation, however, was of a relatively short period and was extremely difficult to control. Both oscillations could be completely eliminated by the addition of a sufficient amount of artificial damping. The control moments produced by the vane-type roll control system were weak and were accompanied by a side force of appreciable magnitude and undesirable direction. In forward flight the model required an undesirably large nose-down tilt angle for equilibrium at any appreciable speed. A vane was placed transversely in the slipstream of the forward duct of the tandem configuration in an attempt to reduce this tilt angle. The vane was effective in reducing the tilt angle but apparently caused an increase in the power requirements and in the angle-of-attack instability. Without the vane, a forward speed of 30 knots (full scale) required a nose-down tilt angle of about 300. A powerful pitch control moment was required not only to maintain the trim attitude but also to overcome the effects of instability with angle of attack. Less pitch control moment was required for the tandem configuration than for the side-by-side configuration at any given forward speed. The instability in roll increased with forward speed. No forward speeds in excess of about 20 knots (full scale) were achieved until the artificial damping in roll and the yaw control moment were increased appreciably above values which had proved satisfactory for hovering flight.

Author

Lateral Control; Scale Models; Ducted Fans; Vertical Landing; Stability; Horizontal Flight

19980228006 NASA Langley Research Center, Hampton, VA USA

Analog-Computer Investigation of Effects of Friction and Preload on the Dynamic Longitudinal Characteristics of a Pilot-Airplane Combination

Crane, Harold L., NASA Langley Research Center, USA; May 1961; 50p; In English

Report No.(s): NASA-TN-D-884; L-1545; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

With an electric analog computer, an investigation has been made of the effects of control frictions and preloads on the transient longitudinal response of a fighter airplane during abrupt small attitude corrections. The simulation included the airplane dynamics, powered control system, feel system, and a simple linearized pseudopilot. Control frictions at the stick pivot and at the servo valve as well as preloads of the stick and valve were considered individually and in combinations. It is believed that the results which are presented in the form of time histories and vector diagrams present a more detailed illustration of the effects of stray forces and compensating forces in the longitudinal control system than has previously been available. Consistent with the results of previous studies, the present results show that any of these four friction and preload forces caused some deterioration of the response. However, even a small amount of valve friction caused an oscillatory pitching response during which the phasing of the valve friction was such that it caused energy to be fed into the pitching oscillation of the air-plane. of the other friction and preload forces which were considered, it was found that stick preload was close to 180 deg. out of phase with valve friction and thus could compensate in large measure for valve friction as long as the cycling of the stick encompassed the trim point. Either stick friction or valve preload provided a smaller stabilizing effect primarily through a reduction in the amplitude of the resultant

force vector acting on the control system. Some data were obtained on the effects of friction when the damping or inertia of the control system or the pilot lag was varied.

Author

Analog Computers; Examination; Friction; Dynamic Characteristics; Loads (Forces); Directional Control; Longitudinal Control

19980228015 NASA Langley Research Center, Hampton, VA USA

Flutter at Very High Speeds

Runyan, Harry L., NASA Langley Research Center, USA; Morgan, Homer G., NASA Langley Research Center, USA; Aug. 1961; 14p; In English

Report No.(s): NASA-TN-D-942; L-1645; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This paper is concerned with a discussion of some of the problems of flutter and aeroelasticity that are or may be important at high speeds. Various theoretical procedures for treating high Mach number flutter are reviewed. Application of two of these methods, namely, the Van Dyke method and piston-theory method, is made to a specific example and compared with linear two- and three-dimensional results. It is shown that the effects of thickness and airfoil shape are destabilizing as compared with linear theory at high Mach number. In order to demonstrate the validity of these large predicted effects, experimental flutter results are shown for two rectangular wings at Mach numbers of 6.86 and 3. The results of nonlinear piston-theory calculations were in good agreement with experiment, whereas the results of using two- and three-dimensional linear theory were not. In addition, some results demonstrating the importance of including camber modes in a flutter analysis are shown, as well as a discussion of one case of flutter due to aerodynamic heating.

Author

Flutter Analysis; Supersonic Speed; Aeroelasticity; Rectangular Wings; Structural Analysis; Aircraft Structures

19980228042 NASA Langley Research Center, Hampton, VA USA

Low-Subsonic Measurements of the Static and Oscillatory Lateral Stability Derivatives of a Sweptback-Wing Airplane Configuration at Angles of Attack from -10 to 90 deg

Hewes, Donald E., NASA Langley Research Center, USA; Jun. 1959; 40p; In English

Report No.(s): NASA-MEMO-5-20-59L; L-365; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been conducted in the Langley free-flight tunnel at low-subsonic speeds to provide some basic information on the stability and control characteristics in the high angle-of-attack range of an airplane configuration typical of current design trends. The investigation consisted of static- and dynamic-force tests over an angle-of-attack range from -10 to 90 deg. The dynamic-force tests, which consisted of both linear- and rotary-oscillation tests, were conducted at values of the reduced-frequency parameter k of 0.10, 0.15, and 0.20. The configuration was directionally unstable for all angles of attack above about 15 deg but maintained positive effective dihedral, control effectiveness, and damping in roll and yaw over most of the angle-of-attack range tested. The effects of frequency on the oscillatory stability derivatives were found to be generally small, but in a few cases the effects were relatively large.

Author

Aerodynamic Configurations; Sweptback Wings; Lateral Stability; Stability Derivatives; Subsonic Speed; Wind Tunnel Tests; Angle of Attack; Free Flight; Lateral Control; Aerodynamic Stability

19980228044 NASA Ames Research Center, Moffett Field, CA USA

The Static and Dynamic Rotary Stability Derivatives at Subsonic Speeds of an Airplane Model Having Wing and Tail Surfaces Swept Back 45 degrees

Lopez, Armando E., NASA Ames Research Center, USA; Buell, Donald A., NASA Ames Research Center, USA; Tinling, Bruce E., NASA Ames Research Center, USA; May 1959; 78p; In English

Report No.(s): NASA-MEMO-5-16-59A; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

Wind-tunnel measurements were made of the static and dynamic rotary stability derivatives of an airplane model having sweptback wing and tail surfaces. The Mach number range of the tests was from 0.23 to 0.94. The components of the model were tested in various combinations so that the separate contribution to the stability derivatives of the component parts and the interference effects could be determined. Estimates of the dynamic rotary derivatives based on some of the simpler existing procedures which utilize static force data were found to be in reasonable agreement with the experimental results at low angles of attack. The

results of the static and dynamic measurements were used to compute the short-period oscillatory characteristics of an airplane geometrically similar to the test model. The results of these calculations are compared with military flying qualities requirements.

Author

Rotary Stability; Dynamic Stability; Static Stability; Subsonic Speed; Aircraft Models; Sweptback Wings; Flight Characteristics

19980228046 NASA Ames Research Center, Moffett Field, CA USA

An Experimental Investigation of the Effect of a Canard Control on the Lift, Drag, and Pitching Moment of an Aspect-Ratio 2.0 Triangular Wing Incorporating a Form of Conical Camber

Menees, Gene P., NASA Ames Research Center, USA; Boyd, John W., NASA Ames Research Center, USA; May 1959; 34p; In English

Report No.(s): NASA-MEMO-5-20-59A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The results of an experimental investigation to determine the effect of a canard control on the lift, drag, and pitching-moment characteristics of an aspect-ratio-2.0 triangular wing incorporating a form of conical camber are presented. The canard had a triangular plan form of aspect ratio 2.0 and was mounted in the extended chord plane of the wing. The ratio of the area of the exposed canard panels to the total wing area was 6.9 percent, and the ratio of the total areas was 12.9 percent. Data were obtained at Mach numbers from 0.70 to 2.22 through an angle-of-attack range from -6 deg to +18 deg with the canard on, and with the canard off. To provide a basis for comparison, the canard was also tested with a symmetrical wing having the same plan form, aspect ratio, and thickness distribution as the cambered wing. The results of the investigation showed that at the high subsonic speeds the gain in maximum lift-drag ratio achieved by camber was considerably reduced by the addition of a canard. At the supersonic speeds, the addition of the canard did not change the effect of camber on the maximum lift-drag ratios.

Author

Lift Drag Ratio; Angle of Attack; Pitching Moments; Delta Wings; Conical Camber; Cambered Wings; Aspect Ratio

19980228049 NASA Ames Research Center, Moffett Field, CA USA

Effects of Large Wing-Tip Masses on Oscillatory Stability of Wing Bending Coupled with Airplane Pitch

Higdon, Donald T., NASA Ames Research Center, USA; Jan. 1959; 34p; In English

Report No.(s): NASA-MEMO-12-29-58A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An examination of oscillatory stability for a straight-winged airplane with large concentrated wing-tip masses was made using wing-bending and airplane-pitching degrees of freedom and considering only quasi-steady aerodynamic forces. It was found that instability caused by coupling of airplane pitching and wing bending occurred for large ratios of effective wing-tip mass to total airplane mass and for coupled wing-bending frequencies near or below the uncoupled pitching frequency. Boundaries for this instability are given in terms of two quantities: (1) the ratio of effective tip mass to airplane mass, which can be estimated, and (2) the ratio of the coupled bending frequency to the uncoupled pitch frequency, which can be measured in flight. These boundaries are presented for various values of several airplane parameters.

Author

Structural Analysis; Aerodynamic Forces; Wings; Bending; Wing Oscillations; Wing Tips; Aircraft Stability; Body-Wing Configurations

19980228050 NASA Ames Research Center, Moffett Field, CA USA

Wind-Tunnel Investigation at Subsonic and Supersonic Speeds of the Static and Dynamic Stability Derivatives of an Airplane Model with an Unswept Wing and a High Horizontal Tail

Lessing, Henry C., NASA Ames Research Center, USA; Butler, James K., NASA Ames Research Center, USA; Jun. 1959; 80p; In English

Report No.(s): NASA-MEMO-6-5-50A; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

Results are presented of a wind-tunnel investigation to evaluate the static and dynamic stability derivatives of a model with a low-aspect-ratio unswept wing and a high horizontal tail. In addition to results for the complete model, results were also obtained of the body alone, body and wing, and body and tail. Data were obtained in the Mach number range from 0.65 to 2.2, at a Reynolds number of 2 million based on the wing mean aerodynamic chord. The angle-of-attack range for most of the data was -11.5 deg to 18 deg. A limited amount of data was obtained with fixed transition. A correspondence between the damping in pitch and the static stability, previously noted in other investigations, was also observed in the present results. The effect observed was that a decrease (or increase) in the static stability was accompanied by an increase (or decrease) in the damping in pitch. A similar correspondence was observed between the damping in yaw and the static-directional stability. Results from similar tests of the same model configuration in two other facilities over different speed ranges are presented for comparison. It was found that most of

the results from the three investigations correlated reasonably well. Estimates of the rotary derivatives were made using available procedures. Comparison with the experimental results indicates the need for development of more precise estimation procedures.

Author

Wind Tunnel Tests; Subsonic Speed; Supersonic Speed; Static Stability; Dynamic Stability; Evaluation

19980228137 NASA Ames Research Center, Moffett Field, CA USA

Estimation of Directional Stability Derivatives at Moderate Angles and Supersonic Speeds

Kaattari, George E., NASA Ames Research Center, USA; Jan. 1959; 74p; In English

Report No.(s): NASA-MEMO-12-1-58A; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

A study of some of the important aerodynamic factors affecting the directional stability of supersonic airplanes is presented. The mutual interference fields between the body, the lifting surfaces, and the stabilizing surfaces are analyzed in detail. Evaluation of these interference fields on an approximate theoretical basis leads to a method for predicting directional stability of supersonic airplanes. Body shape, wing position and plan form, vertical tail position and plan form, and ventral fins are taken into account. Estimates of the effects of these factors are in fair agreement with experiment.

Author

Directional Stability; Estimates; Stability Derivatives; Supersonic Speed

19980228138 NASA Langley Research Center, Hampton, VA USA

Stability Characteristics of Two Missiles of Fineness Ratios 12 and 18 with Six Rectangular Fins of Very Low Aspect Ratio Over a Mach Number Range of 1.4 to 3.2

Henning, Allen B., NASA Langley Research Center, USA; Jan. 1959; 52p; In English

Report No.(s): NASA-MEMO-12-2-58L; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Two rocket-propelled missiles have been test flown by the Langley Pilotless Aircraft Research Division in order to study the stability characteristics of a body with six rectangular fins of very low aspect ratio. The fins, which had exposed aspect ratios of approximately 0.04 and 0.02 per fin, were mounted on bodies of fineness ratios of 12 and 18, respectively. Each body had a nose with a fineness ratio of 3.5 and a cylindrical afterbody. The body and the fin chord of the model having a fineness ratio of 12 were extended the length of 6 body diameters to produce the model with a fineness ratio of 18. The missiles were disturbed in flight by pulse rockets in order to obtain the stability data. The tests were performed over a Mach number range of 1.4 to 3.2 and a Reynolds number range of 2×10^6 to 21×10^6 . The results of these tests indicate that these configurations with the long rectangular fins of very low aspect ratio showed little induced roll with the missile of highest fineness ratio and longest fin chord exhibiting the least amount. Extending the body and fin chord of the shorter missile six body diameters and thereby increasing the fin area approximately 115 percent increased the lift-curve slope based on body cross-sectional area approximately 40 to 55 percent, increased the dynamic stability by a substantial amount, and increased the drag from 14 to 33 percent throughout the comparable Mach number range. The center-of-pressure location of both missiles remained constant over the Mach number range.

Author

Missiles; Pilotless Aircraft; Dynamic Stability; Fineness Ratio; Fins; Low Aspect Ratio

19980228141 NASA Ames Research Center, Moffett Field, CA USA

Stability and Control Characteristics at Subsonic Speeds of a Flat-Top Arrowhead Wing-Body Combination

Buell, Donald A., NASA Ames Research Center, USA; Johnson, Norman S., NASA Ames Research Center, USA; Mar. 1959; 58p; In English

Report No.(s): NASA-MEMO-3-5-59A; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

A wind-tunnel investigation was made to determine the longitudinal- and lateral-stability derivatives of a flat-top wing-body configuration at Mach numbers from 0.22 to 0.90 and Reynolds numbers of 3.5 and 17 million. The wing had a leading-edge sweepback of 78.9 deg and a cathedral of 45 deg on the outer panels. The tests included the determination of the effectiveness of elevon and rudder controls and also an investigation of ground effects. The model was tested at angles of attack up to 28 deg and angles of sideslip up to 18 deg. The dynamic response of this configuration has been determined from the wind-tunnel data for a simulated airplane having a wing loading of 17.7 pounds per square foot. The longitudinal data show a forward shift in aerodynamic center of 10 percent of the mean aerodynamic chord as the lift coefficient is increased above 0.1. Although flown in the lift range of decreasing stability, the simulated airplane did not encounter pitch-up in maneuvers initiated from steady level flight with zero static margin unless a load factor of 2.2 was exceeded. This maneuver margin was provided by a large value of pitching moment due to pitching velocity. The number of cycles to damp the Dutch roll mode to half amplitude, the time constants of the roll subsidence and spiral divergence modes, and control effectiveness in roll are computed. The lateral stability is shown to be positive but is marginal in meeting the military specifications for today's aircraft. An analog computer study has been made in

five degrees of freedom (constant velocity) which illustrates that the handling characteristics are satisfactory. Several programmed rolling maneuvers and coordinated turns also illustrate the handling qualities of the airplane.

Author

Body-Wing Configurations; Aerodynamic Coefficients; Lateral Stability; Longitudinal Stability; Wind Tunnel Tests; Stability Derivatives; Subsonic Speed; Sweptback Wings

19980228144 NASA Langley Research Center, Hampton, VA USA

Investigation of the Stability of Very Flat Spins and Analysis of Effects of Applying various Moments Utilizing the Three Moment Equations of Motion

Klinar, Walter J., NASA Langley Research Center, USA; Grantham, William D., NASA Langley Research Center, USA; Jun. 1959; 50p; In English

Report No.(s): NASA-MEMO-5-25-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Based on linearized equations of motion utilizing only the three moment equations and assuming only flat-spin conditions, it appears that contemporary designs (with the moment of inertia about the wing axis $I_{\text{sub Y}}$) considerably greater than the moment of inertia about the fuselage axis $I_{\text{sub X}}$ having positive values of $C_{\text{sub l, sub p}}$ (rolling-moment coefficient due to rolling) or positive values of $C_{\text{sub l, sub beta}}$ (rolling-moment coefficient due to sideslip) will probably not have a stable spin in the flat-spin region near an angle of attack of 90 deg. If the damping in pitch in flat-spin attitudes is zero, stable flat-spin conditions may not be possible on an airplane having the mass primarily distributed along the wings. The effect of moving ailerons with the spin or the effect of applying a positive pitching moment producing recovery for contemporary fighter designs will be greatest for large negative values of $C_{\text{sub n, sub beta}}$ (yawing-moment coefficient due to sideslip). In addition, for a certain critical value of positive $C_{\text{sub n, sub beta}}$, the rolling moment applied by moving ailerons with the spin or the application of a positive pitching moment will have no effect on reducing the spin rate.

Author

Spin Reduction; Equations of Motion; Yawing Moments; Gyroscopic Stability; Aircraft Stability

19980228212 NASA Langley Research Center, Hampton, VA USA

Effect of Artificial Pitch Damping on the Longitudinal and Rolling Stability of Aircraft with Negative Static Margins

Moul, Martin T., NASA Langley Research Center, USA; Brown, Lawrence W., NASA Langley Research Center, USA; Jun. 1959; 26p; In English

Report No.(s): NASA-MEMO-5-5-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A preliminary theoretical investigation has been made of the short-period longitudinal and steady-rolling (inertia coupling) stability of a hypersonic glider configuration for center-of-gravity locations rear-ward of the airplane neutral point. Such center-of-gravity positions for subsonic flight would improve performance by reducing supersonic and hypersonic static margins and trim drag. Results are presented of stability calculations and a simulator study for a velocity of 700 ft/sec and an altitude of 401,000 feet. With no augmentation, the airplane was rapidly divergent and was considered unsatisfactory in the simulator study. When a pitch damper was employed as a stability augmenter, the short-period mode became overdamped, and the airplane was easily controlled on the simulator. A steady-rolling analysis showed that the airplane can be made free of rolling divergence for all roll rates with an appropriate damper gain.

Author

Numerical Analysis; Hypersonic Gliders; Center of Gravity; Longitudinal Stability

19980228222 NASA Langley Research Center, Hampton, VA USA

Effects of Forebody Deflection on the Stability and Control Characteristics of a Canard Airplane Configuration with a High Trapezoidal Wing at a Mach Number of 2.01

Spearman, M. Leroy, NASA Langley Research Center, USA; Driver, Cornelius, NASA Langley Research Center, USA; Mar. 1959; 30p; In English

Report No.(s): NASA-MEMO-4-4-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been conducted in the Langley 4- by 4-foot supersonic pressure tunnel at a Mach number of 2.01 to determine the effects of forebody deflection on the stability and control characteristics of a canard airplane configuration. The configuration had a high trapezoidal aspect-ratio-3 wing, a trapezoidal canard surface, and a single swept vertical tail. Forebody deflection angles of 0 deg, 2 deg and deg were investigated. The results indicated that nose-up deflections of the forebody provided positive increments of pitching moment with little increase in drag and hence would be useful in reducing the pitch-control requirements and the attendant losses in lift-drag ratio due to trimming. Deflection of the forebody, however, aggravated the decrease in direc-

tional stability with increasing angle of attack by causing a loss in tail contribution and by increasing the instability of the wing-body combination.

Author

Deflection; Forebodies; Stability; Canard Configurations; Trapezoidal Wings

19980228225 NASA Langley Research Center, Hampton, VA USA

Effects of Fuselage Nose Length and a Canopy on the Low-Speed Oscillatory Yawing Derivatives of a Swept-Wing Airplane Model with a Fuselage of Circular Cross Section

Williams, James L., NASA Langley Research Center, USA; DiCamillo, Joseph R., NASA Langley Research Center, USA; Jan. 1959; 30p; In English

Report No.(s): NASA-MEMO-1-15-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A wind-tunnel investigation was made at low speed in the Langley stability tunnel in order to determine the effects of fuselage nose length and a canopy on the oscillatory yawing derivatives of a complete swept-wing model configuration. The changes in nose length caused the fuselage fineness ratio to vary from 6.67 to 9.18. Data were obtained at various frequencies and amplitudes for angles of attack from 0 deg. to about 32 deg. Static lateral and longitudinal stability data are also presented.

Author

Fuselages; Noise (Sound); Wind Tunnel Tests; Length; Oscillations; Static Stability; Aircraft Models

19980228228 NASA Ames Research Center, Moffett Field, CA USA

The Static Longitudinal Stability and Control Characteristics in the Presence of the Ground of a Model Having a Triangular Wing and Canard

Buell, Donald A., NASA Ames Research Center, USA; Tinling, Bruce E., NASA Ames Research Center, USA; Mar. 1959; 34p; In English

Report No.(s): NASA-MEMO-3-4-59A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A wind-tunnel investigation was made of the low-speed characteristics of a canard configuration having triangular wing and canard surfaces with an aspect ratio of 2. The exposed area of the canard was 6.9 percent of the total wing area. The canard hinge line was located at 0.35 of its mean aerodynamic chord and was 0.5 wing mean aerodynamic chord lengths forward of the wing apex. The ground effects, which made the lift more positive and the -Pitching moment more negative at a given angle of attack, were unaffected by the canard. The stability of the model at a constant canard hinge-moment coefficient decreased to 0 near a lift coefficient of 1.0. In addition, the maximum lift coefficient at which the canard could provide balance was decreased by ground effects to less than 1.0 if the moment center was as far forward as 0.21 of the wing mean aerodynamic chord. The relative magnitude of interference effects between the canard and the wing and body is presented.

Author

Longitudinal Stability; Static Stability; Aerodynamic Characteristics; Wind Tunnel Tests

19980228233 NASA Langley Research Center, Hampton, VA USA

Free-Spinning Characteristics of a 1/30-Scale Model of the Grumman WF-2 Airplane

Lee, Henry A., NASA Langley Research Center, USA; 1959; 24p; In English

Report No.(s): NASA-MEMO-4-24-59L; L-326; NASA-AD-3134; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation was conducted in the Langley 20-foot free-spinning tunnel on a 1/30-scale model of the Grumman WF-2 airplane. The effects of control settings and movements upon the erect-spin and recovery characteristics for the flight gross-weight loading with normal center-of-gravity and rearward center-of-gravity positions were determined. For the inverted-spin tests, the flight gross-weight loading with normal center-of-gravity position was used. Brief tests were also made with the radome removed to determine the effect of the radome on the spin and recovery characteristics of the airplane. The results of the tests of the model indicate that erect spins of the airplane in the flight gross-weight loading with the normal (26.3-percent mean aerodynamic chord) center-of-gravity position and with the most rearward (30-percent mean aerodynamic chord) center-of-gravity position possible will be satisfactorily terminated by full rudder reversal to against the spin accompanied by movement of the elevator to at least two-thirds down. With the radome removed, the spin will be steeper and considerably more oscillatory than with the radome on. Recoveries by the preceding technique will be satisfactory. Inverted spins of the airplane will be satisfactorily terminated by full rudder reversal followed by neutralization of the longitudinal and lateral controls.

Author

Wind Tunnel Tests; Scale Models; Spin Tests; Spin Dynamics; Airfoil Profiles; Aerodynamic Configurations; Aircraft Spin; Controllability

19980228235 NASA Langley Research Center, Hampton, VA USA

Flight Investigation of the Stability and Control Characteristics of a 1/4-Scale Model of a Tilt-Wing Vertical-Take-Off-and-Landing Aircraft

Tosti, Louis P., NASA Langley Research Center, USA; Jan. 1959; 24p; In English

Report No.(s): NASA-MEMO-11-4-58L; L-120; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An experimental investigation has been conducted to determine the dynamic stability and control characteristics of a tilt-wing vertical-take-off-and-landing aircraft with the use of a remotely controlled 1/4-scale free-flight model. The model had two propellers with hinged (flapping) blades mounted on the wing which could be tilted up to an incidence angle of nearly 90 deg for vertical take-off and landing. The investigation consisted of hovering flights in still air, vertical take-offs and landings, and slow constant-altitude transitions from hovering to forward flight. The stability and control characteristics of the model were generally satisfactory except for the following characteristics. In hovering flight, the model had an unstable pitching oscillation of relatively long period which the pilots were able to control without artificial stabilization but which could not be considered entirely satisfactory. At very low speeds and angles of wing incidence on the order of 70 deg, the model experienced large nose-up pitching moments which severely limited the allowable center-of-gravity range.

Author

Dynamic Stability; Dynamic Control; Scale Models; Tilt Wing Aircraft; Vertical Landing; Free Flight; Aerodynamic Stability; Wind Tunnel Tests

19980228238 NASA Ames Research Center, Moffett Field, CA USA

Estimation of Static Longitudinal Stability of Aircraft Configurations at High Mach Numbers and at Angles of Attack Between 0 deg and +/-180 deg

Dugan, Duane W., NASA Ames Research Center, USA; Mar. 1959; 76p; In English

Report No.(s): NASA-MEMO-1-17-59A; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

The possibility of obtaining useful estimates of the static longitudinal stability of aircraft flying at high supersonic Mach numbers at angles of attack between 0 and +/-180 deg is explored. Existing theories, empirical formulas, and graphical procedures are employed to estimate the normal-force and pitching-moment characteristics of an example airplane configuration consisting of an ogive-cylinder body, trapezoidal wing, and cruciform trapezoidal tail. Existing wind-tunnel data for this configuration at a Mach number of 6.86 provide an evaluation of the estimates up to an angle of attack of 35 deg. Evaluation at higher angles of attack is afforded by data obtained from wind-tunnel tests made with the same configuration at angles of attack between 30 and 150 deg at five Mach numbers between 2.5 and 3.55. Over the ranges of Mach numbers and angles of attack investigated, predictions of normal force and center-of-pressure locations for the configuration considered agree well with those obtained experimentally, particularly at the higher Mach numbers.

Author

Longitudinal Stability; Static Stability; Aircraft Configurations; Supersonic Speed; Angle of Attack; Trapezoidal Wings; Wind Tunnel Tests

19980228241 NASA Ames Research Center, Moffett Field, CA USA

Static Longitudinal Stability and Control Characteristics of an Unswept Wing and Unswept Horizontal-Tail Configuration at Mach Numbers from 0.70 to 2.22

Peterson, Victor L., NASA Ames Research Center, USA; Menees, Gene P., NASA Ames Research Center, USA; May 1959; 20p; In English

Report No.(s): NASA-MEMO-6-11-59A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Results of an investigation of the static longitudinal stability and control characteristics of an aspect-ratio-3.1, unswept wing configuration equipped with an aspect-ratio-4, unswept horizontal tail are presented without analysis for the Mach number range from 0.70 to 2.22. The hinge line of the all-movable horizontal tail was in the extended wing chord plane, 1.66 wing mean aerodynamic chords behind the reference center of moments. The ratio of the area of the exposed horizontal-tail panels to the total area of the wing was 13.3 percent and the ratio of the total areas was 19.9 percent. Data are presented at angles of attack ranging from -6 deg to +18 deg for the horizontal tail set at angles ranging from +5 deg to -20 deg and for the tail removed.

Author

Longitudinal Stability; Static Stability; Unswept Wings; Tail Assemblies

19980228243 NASA Ames Research Center, Moffett Field, CA USA

The Synthesis of Optimum Homing Missile Guidance Systems with Statistical Inputs

Stewart, Elwood C., NASA Ames Research Center, USA; Smith, Gerald L., NASA Ames Research Center, USA; Apr. 1959; 62p; In English

Report No.(s): NASA-MEMO-2-13-59A; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An analytical approach is presented which is applicable to the optimization of homing navigation guidance systems which are forced to operate in the presence of radar noise. The two primary objectives are to establish theoretical minimum miss distance performance and a method of synthesizing the optimum control system. The factors considered are: (1) target evasive maneuver, (2) radar glint noise, (3) missile maneuverability, and (4) the inherent time-varying character of the kinematics. Two aspects of the problem are considered. In the first, consideration is given only to minimization of the miss distance. The solution given cannot be achieved in practice because the required accelerations are too large. In the second, results are extended to the practical case where the limited acceleration capabilities of the missile are considered by placing a realistic restriction on the mean-square acceleration so that system operation is confined to the linear range. Although the exact analytical solution of the latter problem does not appear feasible, approximate solutions utilizing time-varying control systems can be found. One of these solutions - a range multiplication type control system - is studied in detail. It is shown that the minimum obtainable miss distance with a realistic restriction on acceleration is close to the absolute minimum for unlimited missile maneuverability. Furthermore, it is shown that there is an equivalence in performance between the homing and beam-rider type guidance systems. Consideration is given to the effect of changes in target acceleration, noise magnitude, and missile acceleration on the minimum miss distance.

Author

Numerical Analysis; Guidance (Motion); Missile Control; Optimal Control; Homing; Navigation

19980228269 NASA Langley Research Center, Hampton, VA USA

Flutter Research on Skin Panels

Kordes, Eldon E., NASA Langley Research Center, USA; Tuovila, Weimer J., NASA Langley Research Center, USA; Guy, Lawrence D., NASA Langley Research Center, USA; Sep. 1960; 20p; In English

Report No.(s): NASA-TN-D-451; L-1077; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Representative experimental results are presented to show the current status of the panel flutter problem. Results are presented for unstiffened rectangular panels and for rectangular panels stiffened by corrugated backing. Flutter boundaries are established for all types of panels when considered on the basis of equivalent isotropic plates. The effects of Mach number, differential pressure, and aerodynamic heating on panel flutter are discussed. A flutter analysis of orthotropic panels is presented in the appendix.

Author

Research; Flutter Analysis; Experimentation; Aerodynamic Heating

19980228293 NASA Ames Research Center, Moffett Field, CA USA

Static Stability and Control Characteristics of a 0.5-Scale Model of the Hughes GAR-11 Missile at Mach Numbers from 1.60 to 2.30

Wong, Norman D., NASA Ames Research Center, USA; Ellington, Rex R., NASA Ames Research Center, USA; Jun. 1959; 62p; In English

Report No.(s): NASA-MEMO-6-6-59A; AF-AM-162; A-213; AF-AM-162; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Normal forces, axial forces, pitching moments, and rolling moments on the model and hinge moments on each of the four control surfaces were measured. Control surfaces were deflected from -35 deg to 15 deg in various combinations to produce pitching, yawing, and rolling moments on the model over a range of angles of attack from -5 deg to 25 deg at roll angles from -135 deg to 45 deg.

Author

Rolling Moments; Mach Number; Yawing Moments; Pitching Moments; Control Surfaces

19980228307 NASA Langley Research Center, Hampton, VA USA

Effect of Horizontal-Tail Chord on the Calculated Subsonic Span Loads and Stability Derivatives of Isolated Unswept Tail Assemblies in Sideslip and Steady Roll

Booth, Katherine W., NASA Langley Research Center, USA; Mar. 1959; 34p; In English

Report No.(s): NASA-MEMO-4-1-59L; L-216; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Subsonic span loads and the resulting stability derivatives have been calculated using the discrete-horseshoe-vortex method for a systematic series of horizontal tails in combination with a vertical tail of aspect ratio 1.0 in order to provide information on

the effect of varying the chord of the horizontal tail for isolated tail assemblies performing sideslip and steady-roll motions. In addition, the effects of horizontal-tail dihedral angle for the sideslip case were obtained. Each tail surface considered had a taper ratio of 0.5 and an unswept quarter-chord line. The investigation covered variations in horizontal-tail chord, horizontal-tail span, and vertical location of the horizontal tail. The span loads and the resulting total stability derivatives as well as the vertical- and horizontal-tail contributions to these tail-assembly derivatives are presented in the figures for the purpose of showing the influence of the geometric variables. The results of this investigation showed trends that were in agreement with the results of previous investigations for variations in horizontal-tail span and vertical location of the horizontal tail. Variations in horizontal-tail chord expressed herein in terms of the root-chord ratio, that is, the ratio of horizontal-tail root chord to vertical-tail root chord, were found to have a pronounced influence on most of the span loads and the resulting stability derivatives. For most of the cases considered, the rate of change of the span load coefficients and the stability derivatives with the root-chord ratio was found to be a maximum for small values of root-chord ratio and to decrease as root-chord ratio increased.

Author

Horizontal Tail Surfaces; Horseshoe Vortices; Stability Derivatives; Loads (Forces); Unswept Wings; Roll; Sideslip

19980228321 NASA Langley Research Center, Hampton, VA USA

Ground Simulator Studies of a Nonlinear Linkage in a Power Control System

Assadourian, Arthur, NASA Langley Research Center, USA; Apr. 19549; 20p; In English

Report No.(s): NASA-MEMO-2-15-59L; L-174; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation was made to determine the characteristics of a nonlinear linkage installed in a power control system incorporated in a ground simulator. The nonlinear linkage provided for increased control-stick motion for relatively small simulator response at control motions near neutral. The quality of the control system was rated on the ease and precision with which various tracking tasks were performed by the pilots who operated the simulator. The results obtained with the nonlinear linkage installed in the control system were compared with those obtained by using the normal linear control system. Several combinations of non-linearity of the linkage were tested for various dynamic characteristics of the simulator. It was found that the pilots were able to track almost as well with the nonlinear linkage installed as with the normal system. All of the pilots were of the opinion, however, that the nonlinearity was an undesirable feature in the control system because of the apparent lack of simulator response through the neutral range of the linkage where relatively large stick deflections could be made with very little simulator motion. The results showed that increased lag between the target and chair position, higher stick-force levels, and uneven stick forces due to the dynamics of the linkage were general characteristics of all the nonlinear linkage conditions tested. It was also found that for cases of low simulator damping, rapid control motions caused considerably higher overshoots when the nonlinear linkage was installed than were obtained for the normal linear control system. These characteristics were considered to be sufficiently undesirable to out-weigh the advantages to be gained from the use of a nonlinear linkage in the control system of an airplane.

Author

Dynamic Characteristics; Simulators; Control Sticks; Nonlinearity; Control

19980228345 NASA Langley Research Center, Hampton, VA USA

Longitudinal and Lateral Stability and Control Characteristics of a 1/9-Scale Model of a Twin-Ramjet Canard Target Drone at Mach Numbers from 0.60 to 2.80

Spearman, M. Leroy, NASA Langley Research Center, USA; Aug. 14, 1962; 42p; In English

Report No.(s): NASA-TM-SX-732; L-3030; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Tests were made in the Langley 8-foot transonic pressure tunnel at Mach numbers from 0.60 to 1.20 and in the Langley Unitary Plan tunnel at Mach numbers of 1.60, 1.80, 2.16, and 2.80. Tests were made at combined angles of attack and sideslip with various combinations of control deflections for the model both with and without boosters.

Author

Longitudinal Stability; Lateral Stability; Scale Models; Wind Tunnel Tests; Transonic Speed; Supersonic Speed; Drone Vehicles; Canard Configurations; Ramjet Engines

19980228347 NASA Ames Research Center, Moffett Field, CA USA

A Homing Missile Control System to Reduce the Effects of Radome Diffraction

Smith, Gerald L., NASA Ames Research Center, USA; Oct. 1960; 50p; In English

Report No.(s): NASA-TM-X-395; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The problem of radome diffraction in radar-controlled homing missiles at high speeds and high altitudes is considered from the point of view of developing a control system configuration which will alleviate the deleterious effects of the diffraction. It is shown that radome diffraction is in essence a kinematic feedback of body angular velocities which causes the radar to sense large

apparent line-of-sight angular velocities. The normal control system cannot distinguish between the erroneous and actual line-of-sight rates, and entirely wrong maneuvers are produced which result in large miss distances. The problem is resolved by adding to the control system a special-purpose computer which utilizes measured body angular velocity to extract from the radar output true line-of-sight information for use in steering the missile. The computer operates on the principle of sampling and storing the radar output at instants when the body angular velocity is low and using this stored information for maneuvering commands. In addition, when the angular velocity is not low the computer determines a radome diffraction compensation which is subtracted from the radar output to reduce the error in the sampled information. Analog simulation results for the proposed control system operating in a coplanar (vertical plane) attack indicate a potential decrease in miss distance to an order of magnitude below that for a conventional system. Effects of glint noise, random target maneuvers, initial heading errors, and missile maneuverability are considered in the investigation.

Author

Radar Homing Missiles; Control Systems Design; Diffraction; Radomes; Missile Control; Angular Velocity

19980228349 NASA Dryden Flight Research Center, Edwards, CA USA

Limited Flight Evaluation of Tactair Fluid Control Stability Augmentation Safety System

Jones, Charles K., NASA Dryden Flight Research Center, USA; Jun. 1966; 24p; In English

Report No.(s): NASA-TM-SX-1284; H-438; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A system to provide automatic recovery from high rates of descent and/or airspeed as well as pilot assist during cruise flight has been developed by Tactair Fluid Controls Corporation. A limited evaluation of the system was conducted at the request of the Federal Aviation Agency as a part of an existing program to develop this type of equipment for light aircraft. The automatic activation and recovery performance as well as the autopilot capability were evaluated. Recoveries from, nonextreme attitudes were completed well within the speed and load-factor capability of the aircraft; however, some deficiencies were noted in the operation of the system. As an autopilot, the system offers sufficient stabilization for relief of pilot workload.

Author

Control Stability; Stability Augmentation; Light Aircraft; Flight Tests; Descent; Airspeed; Aircraft Stability; Flight Control

19980228369 NASA Ames Research Center, Moffett Field, CA USA

Estimation of Directional Stability Derivatives at Small Angles and Subsonic and Supersonic Speeds

Goodwin, Frederick K., NASA Ames Research Center, USA; Kaattari, George E., NASA Ames Research Center, USA; Dec. 1958; 96p; In English

Report No.(s): NASA-MEMO-12-2-58A; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

Methods are presented for estimating the directional stability derivative increments contributed by the stabilizing surfaces of subsonic and supersonic aircraft. These methods are strictly applicable at zero angle of attack and small angles of sideslip. The procedure of totaling the incremental coefficients to obtain an estimation of the total empennage side-force and yawing-moment coefficient derivatives is also shown, together with numerical examples. A correlation is presented between estimated and experimental incremental coefficients which indicates that the methods of this report generally estimate the increment of side force gained by the addition of a panel to within +/-10 percent of the experimental value while the yawing-moment increment is generally estimated to within +/-20 percent. This is true for both subsonic and supersonic Mach numbers. An example application of the methods to one of the problems in directional stability, that of minimizing the effect of Mach number on the side-force coefficient derivative of the empennage, is discussed.

Author

Directional Stability; Subsonic Speed; Supersonic Speed; Stability Derivatives; Aircraft Stability

19980228373 NASA Lewis Research Center, Cleveland, OH USA

Preliminary Analysis of the Effect of Flow Separation Due to Rocket Jet Pluming on Aircraft Dynamic Stability During Atmospheric Exit

Dryer, Murray, NASA Lewis Research Center, USA; North, Warren J., NASA Lewis Research Center, USA; Jun. 1959; 38p; In English

Report No.(s): NASA-MEMO-4-22-59E; E-161; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A theoretical investigation was conducted to determine the effects of body boundary-layer separation resulting from a highly underexpanded jet on the dynamic stability of a typical rocket aircraft during an atmospheric exit trajectory. The particular flight condition studied on a digital computer for five degrees of freedom was at Mach 6.0 and 150,000 feet. In view of the unknown character of the separated flow field, two estimates of the pressures in the separated region were made to calculate the unbalanced forces and moments. These estimates, based on limited fundamental zero-angle-of-attack studies and observations, are believed

to cover what may be the actual case. In addition to a fixed control case, two simulated pilot control inputs were studied: rate-limited and instantaneous responses. The resulting motions with and without boundary-layer separation were compared for various initial conditions. The lower of the assumed misalignment forces and moments led to a situation whereby a slowly damped motion could be satisfactorily controlled with rate-limited control input. The higher assumption led to larger amplitude, divergent motions when the same control rates were used. These motions were damped only when the instantaneous control responses were assumed.

Author

Jet Aircraft; Dynamic Stability; Aircraft Stability; Boundary Layer Separation; Evaluation

19980228384 NASA Langley Research Center, Hampton, VA USA

Spin-Entry Characteristics of a Large Supersonic Bomber as Determined by Dynamic Model Tests

Bowman, James S., NASA Langley Research Center, USA; Dec. 1965; 39p; In English

Report No.(s): NASA-TM-SX-1190; L-4604; AF-AM-258; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been conducted in the Langley spin tunnel and at a catapult launch facility of a 1/60-scale dynamic model to determine the spin-entry characteristics of a large supersonic bomber. Catapult tests indicated that spin-entry motions were obtainable for a center-of-gravity location of 0.21 mean aerodynamic chord but were not obtainable at a center-of-gravity location of 0.25 mean aerodynamic chord. Deflected ailerons were effective in promoting or preventing the spin-entry motion and this effect was qualitatively the same as it was for the fully developed spin. Varying the configuration had little significant effect on the spin-entry characteristics. Brief tests conducted with the model in the Langley spin tunnel indicated that fully developed spins were obtainable at the forward center-of-gravity location and that spins were highly unlikely at the rearward center-of-location.

Author

Bomber Aircraft; Airfoil Profiles; Aircraft Spin; Dynamic Models; Scale Models; Catapults; Supersonic Aircraft; Spin Dynamics

19980228392 NASA Langley Research Center, Hampton, VA USA

Transonic Stability and Control Characteristics of a 1/9-Scale Model of a Canard Target Drone Powered by Twin Ramjet Engines

Ayers, Theodore G., NASA Langley Research Center, USA; 1963; 86p; In English

Report No.(s): NASA-TM-SX-806; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

An investigation has been conducted in the Langley 8-foot transonic pressure tunnel to determine the static longitudinal and lateral stability and control characteristics of a 1/9-scale model of a canard target drone powered by twin ramjet engines. The model was tested with and without twin simulated booster rockets, at Mach numbers from 0.60 to 1.20, over a range of angles of attack and sideslip. Both model configurations demonstrate positive static longitudinal and lateral stability and control for the transonic Mach numbers at which they are expected to operate. A substantial difference between the stability levels of the two configurations was noted; this difference might cause some stability and trim problems if it exists at booster separation (Mach number of about 1.60). Differentially extending a pair of off-on spoilers provided adequate roll control at small angles of attack, although adverse yawing moments occurred for the configuration without boosters at positive angles of attack. Reductions in the lateral control effectiveness at high positive and negative angles of attack cause the sideslip angles for which the rolling moments can be trimmed to become very small in some instances. Thus trim problems might be presented if flight should occur under conditions of sideslip.

Author

Static Stability; Canard Configurations; Drone Vehicles; Booster Rocket Engines; Ramjet Engines; Lateral Control; Longitudinal Stability

19980228396 NASA Langley Research Center, Hampton, VA USA

Performance, Stability, and Control Investigation at Mach Numbers from 0.4 to 0.9 of a Model of the "Swallow" with Outer Wing Panels Swept 25 degree with and without Power Simulation

Runckel, Jack F., NASA Langley Research Center, USA; Schmeer, James W., NASA Langley Research Center, USA; Cassetti, Marlowe D., NASA Langley Research Center, USA; May 03, 1960; 70p; In English

Report No.(s): NASA-TM-SX-296; L-975; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An investigation of the performance, stability, and control characteristics of a variable-sweep arrow-wing model (the "Swallow") with the outer wing panels swept 25 deg has been conducted in the Langley 16-foot transonic tunnel. The wing was uncambered and untwisted and had RAE 102 airfoil sections with a thickness-to-chord ratio of 0.14 normal to the leading edge. Four outboard engines located above and below the wing provided propulsive thrust, and, by deflecting in the pitch direction and rotating in the lateral plane, also produced control forces. A pair of swept lateral fins and a single vertical fin were mounted on each engine nacelle to provide aerodynamic stability and control. Jets-off data were obtained with flow-through nacelles, stimulating the effects of inlet flow; jet thrust and hot-jet interference effects were obtained with faired-nose nacelles housing hydrogen perox-

ide gas generators. Six-component force and moment data were obtained through a Mach number range of 0.40 to 0.90 at angles of attack and angles of sideslip from 0 deg to 15 deg. Longitudinal, directional, and lateral control were obtained by deflecting the nacelle-fin combinations as elevators, rudders, and ailerons at several fixed angles for each control.

Author

Aerodynamic Stability; Stability; Wing Panels; Control Surfaces; Directional Control; Inlet Flow; Longitudinal Control; Lateral Control

19980228401 NASA Ames Research Center, Moffett Field, CA USA

Pitch-Up Problem: A Criterion and Method of Evaluation

Sadoff, Melvin, NASA Ames Research Center, USA; Feb. 1959; 18p; In English

Report No.(s): NASA-MEMO-3-7-59A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A method has been described for predicting the probable relative severity of pitch-up of a new airplane design prior to initial flight tests. An illustrative example has been presented which demonstrated the use of this procedure for evaluating the pitch-up behavior of a large, relatively flexible airplane. It has also been shown that for airplanes for which a mild pitch-up tendency is predicted, the wing and tail loads likely to be encountered in pitch-up maneuvers would not assume critical values, even for pilots unfamiliar with pitch-up.

Author

Aircraft Design; Flight Tests; Pitching Moments; Wing Loading; Aircraft Stability

19980228402 NASA Langley Research Center, Hampton, VA USA

Force-Test Investigation of the Stability and Control Characteristics of a 1/4-Scale Model of a Tilt-Wing Vertical-Take-Off-and-Landing Aircraft

Newsom, William A., Jr., NASA Langley Research Center, USA; Tosti, Louis P., NASA Langley Research Center, USA; Jan. 1959; 58p; In English

Report No.(s): NASA-MEMO-11-3-58L; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

A wind-tunnel investigation has been made to determine the aerodynamic characteristics of a 1/4-scale model of a tilt-wing vertical-take-off-and-landing aircraft. The model had two 3-blade single-rotation propellers with hinged (flapping) blades mounted on the wing, which could be tilted from an incidence of 4 deg for forward flight to 86 deg for hovering flight. The investigation included measurements of both the longitudinal and lateral stability and control characteristics in both the normal forward flight and the transition ranges. Tests in the forward-flight condition were made for several values of thrust coefficient, and tests in the transition condition were made at several values of wing incidence with the power varied to cover a range of flight conditions from forward-acceleration (or climb) conditions to deceleration (or descent) conditions. The control effectiveness of the all-movable horizontal tail, the ailerons and the differential propeller pitch control was also determined. The data are presented without analysis.

Author

Aerodynamic Characteristics; Longitudinal Stability; Lateral Stability; Flight Conditions; Scale Models; Wind Tunnel Tests; Flapping; Vertical Landing

19980228405 NASA Ames Research Center, Moffett Field, CA USA

Sampled-Data Techniques Applied to a Digital Controller for an Altitude Autopilot

Schmidt, Stanley F., NASA Ames Research Center, USA; Harper, Eleanor V., NASA Ames Research Center, USA; Jun. 1959; 76p; In English

Report No.(s): NASA-MEMO-4-14-59A; A-138; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

Sampled-data theory, using the Z transformation, is applied to the design of a digital controller for an aircraft-altitude autopilot. Particular attention is focused on the sensitivity of the design to parameter variations and the abruptness of the response, that is, the normal acceleration required to carry out a transient maneuver. Consideration of these two characteristics of the system has shown that the finite settling time design method produces an unacceptable system, primarily because of the high sensitivity of the response to parameter variations, although abruptness can be controlled by increasing the sampling period. Also demonstrated is the importance of having well-damped poles or zeros if cancellation is attempted in the design methods. A different method of smoothing the response and obtaining a design which is not excessively sensitive is proposed, and examples are carried through to demonstrate the validity of the procedure. This method is based on design concepts of continuous systems, and it is shown that if no pole-zero cancellations are allowed in the design, one can obtain a response which is not too abrupt, is relatively insensitive to parameter variations, and is not sensitive to practical limits on control-surface rate. This particular design also has

the simplest possible pulse transfer function for the digital controller. Simulation techniques and root loci are used for the verification of the design philosophy.

Author

Data Flow Analysis; Digital Systems; Automatic Pilots; Altitude Control; Design Analysis; Sampled Data Systems; Control Systems Design

19980228407 NASA Langley Research Center, Hampton, VA USA

Some Effects of Yaw Damping on Airplane Motions and Vertical-Tail Loads in Turbulent Air

Funk, Jack, NASA Langley Research Center, USA; Cooney, T. V., NASA Langley Research Center, USA; Mar. 1959; 10p; In English

Report No.(s): NASA-MEMO-2-17-59L; L-433; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

Results of analytical and flight studies are presented to indicate the effect of yaw damping on the airplane motions and the vertical-tail loads in rough air. The analytical studies indicate a rapid reduction in loads on the vertical tail as the damping is increased up to the point of damping the lateral motions to 1/2 amplitude in one cycle. Little reduction in load is obtained by increasing the lateral damping beyond that point. Flight measurements made in rough air at 5,000 and 35,000 feet on a large swept-wing bomber equipped with a yaw damper show that the yaw damper decreased the loads on the vertical tail by about 50 percent at 35,000 feet. The reduction in load at 5,000 feet was not nearly as great. Measurements of the pilot's ability to damp the lateral motions showed that the pilot could provide a significant amount of damping but that manual control was not as effective as a yaw damper in reducing the loads.

Author

Yaw; Damping; In-Flight Monitoring; Tail Assemblies; Measuring Instruments

19980228451 NASA Langley Research Center, Hampton, VA USA

Investigation of the Subsonic Stability and Control Characteristics of a 1/7-Scale Model of the North American X-15 Airplane with and without Fuselage Forebody Strakes

Hassell, James L., Jr., NASA Langley Research Center, USA; Hewes, Donald E., NASA Langley Research Center, USA; Feb. 1960; 48p; In English

Report No.(s): NASA-TM-X-210; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation of the low-subsonic stability and control characteristics of a 1/7-scale free-flying model modified to represent closely the North American X-15 airplane (configuration 3) has been made in the Langley full-scale tunnel. Flight conditions at a relatively low altitude were simulated with the center of gravity at 16.0 percent of the mean aerodynamic chord. The longitudinal stability and control were considered to be satisfactory for all flight conditions tested. The lateral flight behavior was generally satisfactory for angles of attack below about 20 deg. At higher angles, however, the model developed a tendency to fly in a side-slipped attitude because of static directional instability at small sideslip angles. Good roll control was maintained to the highest angles tested, but rudder effectiveness diminished with increasing angle of attack and became adverse for angles above 40 deg. Removal of the lower rudder had little effect on the lateral flight characteristics for angles of attack less than about 20 deg but caused the lateral flight behavior to become worse in the high angle-of-attack range. The addition of small fuselage forebody strakes improved the static directional stability and lateral flight behavior of both configurations.

Author

Scale Models; Airfoil Profiles; Flight Characteristics; Flight Conditions; Longitudinal Stability; Static Stability; Lateral Control; Directional Stability

19980228460 NASA Langley Research Center, Hampton, VA USA

Problems Involved in an Emergency Method of Guiding a Gliding Vehicle from High Altitudes to a High Key Position

Jewel, Joseph W., Jr., NASA Langley Research Center, USA; Whitten, James B., NASA Langley Research Center, USA; Aug. 1960; 24p; In English

Report No.(s): NASA-TN-D-438; L-1063; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been conducted to determine the problems involved in an emergency method of guiding a gliding vehicle from high altitudes to a high key position (initial position) above a landing field. A jet airplane in a simulated flameout condition, conventional ground-tracking radar, and a scaled wire for guidance programming on the radar plotting board were used in the tests. Starting test altitudes varied from 30,000 feet to 46,500 feet, and starting positions ranged 8.4 to 67 nautical miles from the high key. Specified altitudes of the high key were 12,000, 10,000 or 4,000 feet. Lift-drag ratios of the aircraft of either 17, 16, or 6 were held constant during any given flight; however, for a few flights the lift-drag ratio was varied from 11 to 6. Indicated airspeeds were held constant at either 160 or 250 knots. Results from these tests indicate that a gliding vehicle having a lift-drag ratio of

16 and an indicated approach speed of 160 knots can be guided to within 800 feet vertically and 2,400 feet laterally of a high key position. When the lift-drag ratio of the vehicle is reduced to 6 and the indicated approach speed is raised to 250 knots, the radar controller was able to guide the vehicle to within 2,400 feet vertically and 2,400 feet laterally of the high key. It was also found that radar stations which give only azimuth-distance information could control the glide path of a gliding vehicle as well as stations that receive azimuth-distance-altitude information, provided that altitude information is supplied by the pilot.

Author

Aerodynamic Drag; Glide Paths; High Altitude; Lift Drag Ratio; Controllers

19980228468 NASA Langley Research Center, Hampton, VA USA

Low-Subsonic Static Stability and Damping Derivatives at Angles of Attack From 0 deg to 90 deg for a Model with a Low-Aspect-Ratio Unswept Wing and Two Different Fuselage Forebodies

Boisseau, Peter C., NASA Langley Research Center, USA; Mar. 1959; 28p; In English

Report No.(s): NASA-MEMO-1-22-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been made in the Langley free-flight tunnel at low-subsonic speed to determine the static stability, control effectiveness, and damping in roll and yaw of a model with a low-aspect-ratio unswept wing and two different fuselage forebodies at angles of attack from 0 deg to 90 deg. Results were obtained with a fuselage configuration having a long pointed nose and a shorter rounded nose. Although the wing stalled at an angle of attack of about 12 deg, maximum lift did not occur until an angle of attack of about 40 deg or 50 deg was obtained. The static longitudinal stability of the model having a short rounded nose was greater than that of the model having a longer pointed nose over the entire angle-of-attack range. The pointed-nose model had large out-of-trim yawing moments above an angle of attack of about 40 deg. Shortening and rounding the nose of the model delayed these out-of-trim yawing moments to slightly higher angles of attack. Both models were directionally unstable above an angle of attack of about 20 deg, but both had positive effective dihedral over virtually the entire angle-of-attack range. At the higher angles of attack the pointed-nose model had generally better damping in roll than that of the rounded-nose model. Both models had very high damping in yaw at an angle of attack of about 50 deg or 60 deg.

Author

Low Aspect Ratio Wings; Forebodies; Unswept Wings; Fuselages; Angle of Attack; Controllability

19980228475 NASA Langley Research Center, Hampton, VA USA

Analytical Investigation of a Flicker-Type Roll Control for a Mach Number 6 Missile with Aerodynamic Controls Over An Altitude Range of 82,000 to 282,000 feet

Lundstrom, Reginald R., NASA Langley Research Center, USA; Whitman, Ruth I., NASA Langley Research Center, USA; May 1959; 52p; In English

Report No.(s): NASA-MEMO-4-23-59L; L-211; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An analytical investigation has been carried out to determine the responses of a flicker-type roll control incorporated in a missile which traverses a range of Mach number of 6.3 at an altitude of 82,000 feet to 5.26 at an altitude of 282,000 feet. The missile has 80 deg delta wings in a cruciform arrangement with aerodynamic controls attached to the fuselage near the wing trailing edge and indexed 450 to the wings. Most of the investigation was carried out on an analog computer. Results showed that roll stabilization that may be adequate for many cases can be obtained over the altitude range considered, providing that the rate factor can be changed with altitude. The response would be improved if the control deflection were made larger at the higher altitudes. lag times less than 0.04 second improve the response appreciably. Asymmetries that produce steady rolling moments can be very detrimental to the response in some cases. The wing damping made a negligible contribution to the response.

Author

Control Surfaces; Mach Number; Flicker; Rolling Moments; Analog Computers

19980230602 NASA Langley Research Center, Hampton, VA USA

Low-Speed Static Stability and Control Characteristics of a Model of a Right Triangular Pyramid Reentry Configuration

Paulson, John W., NASA Langley Research Center, USA; Apr. 1959; 18p; In English

Report No.(s): NASA-MEMO-4-11-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation of the low-speed static stability and control characteristics of a model of a right triangular pyramid reentry configuration has been made in the Langley free-flight tunnel. The investigation showed that the model had generally satisfactory longitudinal and lateral static stability characteristics. The maximum lift-drag ratio was increased from about 3 to 5 by boattailing the base of the model.

Author

Free Flight; Lift Drag Ratio; Longitudinal Stability; Static Stability

19980230615 NASA, Washington, DC USA

The Airplane as an Object of Control: Block Diagrams of the Equations of Perturbed Airplane Motion

Vedrov, V. S.; Romanov, G. L.; Surina, V. N.; Ministry of Aviation Industry; Oct. 1959, No. 74; 48p; In English; Translated under NASA contract by Consultants Custom Translations, Inc., New York, NY

Report No.(s): NASA-TT-F-5; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This work considers the presentation of equations for the perturbed motion of an airplane in the form of single-loop block diagrams. A brief analysis is given of the characteristics of the individual links and of their change with changing flight speed and altitude. A derivation of transfer functions for control with elevator, rudder, and aileron is presented, as well as simplified expressions for the transfer functions, depending on the frequency range, which correspond to breaking up the perturbed motion into simple types. The representation of the equations of perturbed airplane motion in the form of single-loop block diagrams permits a simple and easy application of contemporary methods of control theory to the analysis of airplane motion, and also allows rapid formulation of simplified equations of motion and transfer functions applicable during control with the control elements.

Author

Control Theory; Equations of Motion; Aircraft Stability

19980230618 NASA Ames Research Center, Moffett Field, CA USA

The Effect of Lateral- and Longitudinal- Range Control on Allowable Entry Conditions for a Point Return from Space

Boissevain, Alfred G., NASA Ames Research Center, USA; Jul. 1961; 14p; In English

Report No.(s): NASA-TN-D-1067; A-506; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The problem of return to a specified landing point on the earth from flight in space is considered by studying the interaction between an assumed control over the lateral and longitudinal range and the initial conditions of approach to the earth, given by orbital-plane inclination, vacuum perigee location, and time of arrival. The maneuvering capability in the atmosphere permits a point return for a range of entry conditions. A lateral-range capability of ± 500 miles from the center line of an entry trajectory can allow a variation in the time of arrival of over 3.5 hours. Variation in the orbital-plane inclination angle can be as much as ± 13 deg.

Author

Orbital Mechanics; Earth Orbits; Earth Orbital Rendezvous; Trajectory Control; Perigees

19980230619 NASA Langley Research Center, Hampton, VA USA

Low-Speed Measurements of Static and Oscillatory Lateral Stability Derivatives of a 1/5-scale Model of a Jet-Powered Vertical-Attitude VTOL Research Airplane

Shanks, Robert E., NASA Langley Research Center, USA; Smith, Charles C., Jr., NASA Langley Research Center, USA; Sep. 1960; 22p; In English

Report No.(s): NASA-TN-D-433; L-640; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been made in the Langley free-flight tunnel to determine the low-speed static lateral stability characteristics and the rolling, yawing, and sideslipping dynamic stability derivatives of a 1/5-scale model of a jet-powered vertical-attitude VTOL research airplane. The results of this investigation are presented herein without analysis.

Author

Vertical Takeoff Aircraft; Aerodynamic Coefficients; Aerodynamic Stability; Aerodynamic Characteristics; Attitude (Inclination); Dynamic Stability

19980230678 NASA Ames Research Center, Moffett Field, CA USA

A Buffet Investigation at High Subsonic Speeds of Wing-Fuselage-Tail Combinations having Sweptback Wings with NACA Four-Digit Thickness Distributions, Fences, and Body Contouring

Sutton, Fred B., NASA Ames Research Center, USA; Mar. 1959; 56p; In English

Report No.(s): NASA-MEMO-3-23-59A; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An investigation has been made to determine the effect of wing fences, fuselage contouring, varying wing sweepback angle from 40 deg. to 45 deg., mounting the horizontal tail on an outboard boom) and wing thickness distribution upon the buffeting response of typical airplane configurations employing sweptback wings of high aspect ratio. The tests were conducted through an angle-of-attack range at Mach numbers varying from 0.60 to 0.92 at a Reynolds number of 2 million. For the combinations with 40 deg. of sweepback, the addition of multiple wing fences usually decreased the buffeting at moderate and high lift coefficients and reduced the erratic variation of buffet intensities with increasing lift coefficient and Mach number. Fuselage contouring also reduced buffeting but was not as effective as the wing fences. At most Mach numbers, buffeting occurred at higher lift coefficients for the combination with the NACA 64A thickness distributions than for the combination with the NACA four-digit thick-

ness distributions. At high subsonic speeds, heavy buffeting was usually indicated at lift coefficients which were lower than the lift coefficients for static-longitudinal instability. The addition of wing fences improved the pitching-moment characteristics but had little effect on the onset of buffeting. For most test conditions and model configurations, the root-mean-square and the maximum values measured for relative buffeting indicated similar effects and trends; however, the maximum buffeting loads were usually two to three times the root-mean-square intensities.

Author

Aerodynamic Coefficients; Angle of Attack; High Aspect Ratio; Mach Number; Pitching Moments; Reynolds Number; Sweptback Wings; Thickness; Wings

19980230679 NASA Langley Research Center, Hampton, VA USA

Flutter Tests of Some Simple Models at a Mach Number of 7.2 in Helium Flow

Morgan, Homer G., NASA Langley Research Center, USA; Miller, Robert W., NASA Langley Research Center, USA; Apr. 1959; 30p; In English

Report No.(s): NASA-MEMO-4-8-59L; L-199; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Results of hypersonic flutter tests on some simple models are presented. The models had rectangular plan forms of panel aspect ratio 1.0, no sweepback, and bending-to-torsion frequency ratios of about 1/3. Two airfoil sections were included in the tests; double wedges of 5-, 10-, and 15-percent thickness and flat plates with straight, parallel sides and beveled leading and trailing edges. The models were supported by a cantilevered shaft. The double-wedge wings were tested in helium at a Mach number of 7.2. An effect of airfoil thickness on flutter speed was found, thicker wings requiring more stiffness to avoid flutter. A few tests in air at a Mach number of 6.9 showed the same thickness effect and also indicated that tests in helium would predict conservative flutter boundaries in air. The data in air and helium seemed to be correlated by piston-theory calculations. Piston-theory calculations agreed well with experiment for the thinner models but began to deviate as the thickness parameter MT approached and exceeded 1.0. A few tests on flat-plate models with various elastic-axis locations were made. Piston-theory calculations would not satisfactorily predict the flutter of these models, probably because of their blunt leading edges.

Author

Flutter; Performance Tests; Models; Helium; Gas Flow; Hypersonic Speed

19980231064 NASA Langley Research Center, Hampton, VA USA

Low-Speed Static Stability Characteristics of Two Configurations Suitable for Lifting Reentry from Satellite Orbit

Paulson, John W., NASA Langley Research Center, USA; Nov. 1958; 20p; In English

Report No.(s): NASA-MEMO-10-22-58L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation of the low-speed static stability and control characteristics of 1/4-scale models of two configurations suitable for lifting reentry from satellite orbit has been made in the Langley free-flight tunnel. One of the models was a thick, all-wing configuration having a delta plan form and the other was a flat delta wing with a half-cone fuselage. The investigation showed that, in general, the all-wing configuration had better longitudinal and lateral stability characteristics than the flat delta configuration.

Author

Low Speed; Static Stability; Aerodynamic Configurations; Scale Models; Experimentation; Wind Tunnel Tests

19980231089 NASA Langley Research Center, Hampton, VA USA

Flight Investigation of an Automatic Throttle Control in Landing Approaches

Lina, Lindsay J., NASA Langley Research Center, USA; Champine, Robert A., NASA Langley Research Center, USA; Morris, Garland J., NASA Langley Research Center, USA; Mar. 1959; 10p; In English

Report No.(s): NASA-MEMO-2-19-59L; L-432; No Copyright; Avail: CASI; A02, Hardcopy; A01, Microfiche

A flight investigation of an automatic throttle control in landing approaches has been made. It was found that airspeed could be maintained satisfactorily by the automatic throttle control. Turbulent air caused undesirably large variations of engine power which were uncomfortable and disconcerting; nevertheless, the pilot felt that he could make approaches 5 knots slower with equal assurance when the automatic control was in operation.

Author

Automatic Control; Throttling; Flight Tests; Landing

09
RESEARCH AND SUPPORT FACILITIES (AIR)

Includes airports, hangars and runways; aircraft repair and overhaul facilities; wind tunnels; shock tubes; and aircraft engine test stands.

19980228143 NASA Ames Research Center, Moffett Field, CA USA

Use of Flight Simulators for Pilot-Control Problems

Rathert, George A., Jr., NASA Ames Research Center, USA; Creer, Brent Y., NASA Ames Research Center, USA; Douvillier, Joseph G., Jr., NASA Ames Research Center, USA; Feb. 1959; 18p; In English

Report No.(s): NASA-MEMO-3-6-59A; A-243; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Comparisons have been made between actual flight results and results obtained with fixed and moving flight simulators in a number of phases of flying airplanes with a wide range of characteristics. These results have been used to study the importance of providing motion stimuli in a simulator in order that the pilot operate the simulator in a realistic manner. Regions of airplane characteristics where motion stimuli are either mandatory or desirable are indicated.

Author

Flight Simulators; Pilot Training; Motion Simulation

19980228216 NASA Lewis Research Center, Cleveland, OH USA

Pressure Drag of Axisymmetric Cowls Having Large Initial Lip Angles at Mach Numbers from 1.90 to 4.90

Samanich, Nick E., NASA Lewis Research Center, USA; Jan. 1959; 18p; In English

Report No.(s): NASA-MEMO-1-10-59E; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The results of experimental and theoretical data on nine cowls are presented to determine the effect of initial lip angle and projected frontal area on the cowl pressure drag coefficient at Mach numbers from 1.90 to 4.90. The experimental drag coefficients were approximated well with two-dimensional shock-expansion theory at the lower cowl-projected areas, but the difference between theory and experiment increased as the cowl area ratio was increased or as shock detachment at the cowl lips was approached. An empirical chart is presented, which can be used to estimate the cowl pressure drag coefficient of cowls approaching an elliptic contour.

Author

Pressure Drag; Cowlings; Axisymmetric Flow

19980228217 NASA Langley Research Center, Hampton, VA USA

A Hydrogen Peroxide Hot-Jet Simulator for Wind-Tunnel Tests of Turbojet-Exit Models

Runckel, Jack F., NASA Langley Research Center, USA; Swihart, John M., NASA Langley Research Center, USA; Feb. 1959; 38p; In English

Report No.(s): NASA-MEMO-1-10-59L; L-110; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A turbojet-engine-exhaust simulator which utilizes a hydrogen peroxide gas generator has been developed for powered-model testing in wind tunnels with air exchange. Catalytic decomposition of concentrated hydrogen peroxide provides a convenient and easily controlled method of providing a hot jet with characteristics that correspond closely to the jet of a gas turbine engine. The problems associated with simulation of jet exhausts in a transonic wind tunnel which led to the selection of a liquid monopropellant are discussed. The operation of the jet simulator consisting of a thrust balance, gas generator, exit nozzle, and auxiliary control system is described. Static-test data obtained with convergent nozzles are presented and shown to be in good agreement with ideal calculated values.

Author

Hydrogen; Jet Exhaust; Simulators; Fabrication; Gas Generators; Wind Tunnel Models; Powered Models; Convergent Nozzles

10
ASTRONAUTICS

Includes astronautics (general); astrodynamics; ground support systems and facilities (space); launch vehicles and space vehicles; space transportation; space communications, spacecraft communications, command and tracking; spacecraft design, testing and performance; spacecraft instrumentation; and spacecraft propulsion and power.

19980227819 NASA Langley Research Center, Hampton, VA USA

Analytical Investigation of the Dynamic Behavior of a Nonlifting Manned Reentry Vehicle

Lichtenstein, Jacob H., NASA Langley Research Center, USA; Sep. 1960; 58p; In English

Report No.(s): NASA-TN-D-416; L-867; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An analytic investigation was made of the dynamic behavior of a nonlifting manned reentry vehicle as it descended through the atmosphere. The investigation included the effects of variations in the aerodynamic stability derivatives, the spin rate, reentry angle, and velocity. The effect of geostrophic winds and of employing a drogue parachute for stability purposes were also investigated. It was found that for the portion of the flight above a Mach number of 1 a moderate amount of negative damping could be tolerated but below a Mach number of 1 good damping is necessary. The low-speed stability could be improved by employing a drogue parachute. The effectiveness of the drogue parachute was increased when attached around the periphery of the rear of the vehicle rather than at the center. Neither moderate amounts of spin or the geostrophic winds had appreciable effects on the stability of the vehicle. The geostrophic winds and the reentry angle or velocity all showed important effects on the range covered by the reentry flight path.

Author

Aerodynamic Stability; Reentry Vehicles; Stability Derivatives; Manned Reentry; Geostrophic Wind; Dynamic Characteristics

19980227837 NASA Ames Research Center, Moffett Field, CA USA

Point Return from a Lunar Mission for a Vehicle that Maneuvers Within the Earth's Atmosphere

Sommer, Simon C., NASA Ames Research Center, USA; Short, Barbara J., NASA Ames Research Center, USA; Nov. 1961; 34p; In English

Report No.(s): NASA-TN-D-1142; A-553; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been made of point return of a vehicle with a lift-to-drag ratio of 1/2, returning from a lunar mission. It was found that the available longitudinal and lateral range allowed considerable tolerances in entry conditions for a point return. Longitudinal range capability for a vehicle that was allowed to skip to an altitude not exceeding 400 miles was about 3-1/2 times greater than the range capability of a vehicle that was restricted to remain in the atmosphere after entry. Longitudinal range is very sensitive to changes in both velocity and flight-path angle at the bottom of the first pull-out and at exit. An investigation showed that after a skip a vehicle could be placed in a circular orbit for a relatively modest weight penalty. A skip maneuver was found to have no effect on lateral range when the roll was initiated at a velocity near satellite speed after the vehicle had re-entered the atmosphere. However, when the roll was initiated at the earliest possible time along the undershoot boundary, lateral range was increased by a factor of about 2-1/2. The tolerable errors in time of arrival and in inclination of the orbital plane at point of entry were greater for the skip trajectory than for the no-skip trajectory.

Author

Space Missions; Circular Orbits; Lift Drag Ratio; Earth Atmosphere; Flight Paths

19980227845 NASA Langley Research Center, Hampton, VA USA

Effects of Booster Transition Section and Rudder Deflection on the Low-Angle-of-Attack Static Stability of a Winged Reentry Vehicle at Mach Numbers of 10.8 and 17.8 in Helium

Ladson, Charles L., NASA Langley Research Center, USA; Jan. 1962; 28p; In English

Report No.(s): NASA-TM-X-624; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation has been carried out to obtain aerodynamic stability and control data on a model of a winged reentry vehicle at Mach numbers of 10.8 and 17.8 in helium. The effects of a booster transition section on the static stability were obtained at angles of attack from -5 deg to 15 deg and at angles of sideslip from -5 deg to 10 deg at an angle of attack of 0 deg. Directional control data were also obtained at an angle of attack of 0 deg for sideslip angles from -5 deg to 10 deg. No detailed analysis of the data has been made.

Author

Reentry Vehicles; Aerodynamic Stability; Angle of Attack; Rudders; Static Stability; Directional Control

19980227963 NASA Langley Research Center, Hampton, VA USA

Noise Considerations for Manned Reentry Vehicles

Hilton, David A., NASA Langley Research Center, USA; Mayes, William H., NASA Langley Research Center, USA; Hubbard, Harvey H., NASA Langley Research Center, USA; Sep. 1960; 18p; In English

Report No.(s): NASA-TN-D-450; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Noise measurements pertaining mainly to the static firing, launch, 0 and exit flight phases are presented for three rocket-powered vehicles 4 in the Project Mercury test program. Both internal and external data 4 from onboard recordings are presented for a range of Mach numbers and dynamic pressures and for different external vehicle shapes. The main sources of noise are noted to be the rocket engines during static firing and launch and the aerodynamic boundary layer during the high-dynamic-pressure portions of the flight. Rocket-engine noise measurements along the surface of the Mercury Big Joe vehicle were noted to correlate well with data from small models and available data for other large rockets. Measurements have indicated that the aerodynamic noise pressures increase approximately as the dynamic pressure increases and may vary according to the external shape of the vehicle, the highest noise levels being associated with conditions of flow separation. There is also a trend for the aerodynamic noise spectra to peak at higher frequencies as the flight Mach number increases.

Author

Reentry Vehicles; Manned Reentry; Rocket Vehicles; Rocket Engine Noise; Noise Measurement; Dynamic Pressure; Aerodynamic Noise

19980227974 NASA Ames Research Center, Moffett Field, CA USA

Study of a Satellite Attitude Control System Using Integrating Gyros as Torque Sources

White, John S., NASA Ames Research Center, USA; Hansen, Q. Marion, NASA Ames Research Center, USA; Sep. 1961; 42p; In English

Report No.(s): NASA-TN-D-1073; A-443; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This report considers the use of single-degree-of-freedom integrating gyros as torque sources for precise control of satellite attitude. Some general design criteria are derived and applied to the specific example of the Orbiting Astronomical Observatory. The results of the analytical design are compared with the results of an analog computer study and also with experimental results from a low-friction platform. The steady-state and transient behavior of the system, as determined by the analysis, by the analog study, and by the experimental platform agreed quite well. The results of this study show that systems using integrating gyros for precise satellite attitude control can be designed to have a reasonably rapid and well-damped transient response, as well as very small steady-state errors. Furthermore, it is shown that the gyros act as rate sensors, as well as torque sources, so that no rate stabilization networks are required, and when no error sensor is available, the vehicle is still rate stabilized. Hence, it is shown that a major advantage of a gyro control system is that when the target is occulted, an alternate reference is not required.

Author

Satellite Attitude Control; Gyroscopes; Torque; Satellite Instruments; Control Systems Design; Design Analysis; Flight Instruments

19980228012 NASA Langley Research Center, Hampton, VA USA

Exploratory Environmental Tests of Several Heat Shields

Goodman, George P., NASA Langley Research Center, USA; Betts, John, Jr., NASA Langley Research Center, USA; Sep. 1961; 54p; In English

Report No.(s): NASA-TN-D-897; L-1524; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Exploratory tests have been conducted with several conceptual radiative heat shields of composite construction. Measured transient temperature distributions were obtained for a graphite heat shield without insulation and with three types of insulating materials, and for a metal multipost heat shield, at surface temperatures of approximately 2,000 F and 1,450 F, respectively, by use of a radiant-heat facility. The graphite configurations suffered loss of surface material under repeated irradiation. Temperature distribution calculated for the metal heat shield by a numerical procedure was in good agreement with measured data. Environmental survival tests of the graphite heat shield without insulation, an insulated multipost heat shield, and a stainless-steel-tile heat shield were made at temperatures of 2,000 F and dynamic pressures of approximately 6,000 lb/sq ft, provided by an ethylene-heated jet operating at a Mach number of 2.0 and sea-level conditions. The graphite heat shield survived the simulated aerodynamic heating and pressure loading. A problem area exists in the design and materials for heat-resistant fasteners between the graphite shield and the base structure. The insulated multipost heat shield was found to be superior to the stainless-steel-tile heat shield in retarding heat flow. Over-lapped face-plate joints and surface smoothness of the insulated multi- post heat shield were

not adversely affected by the test environment. The graphite heat shield without insulation survived tests made in the acoustic environment of a large air jet. This acoustic environment is random in frequency and has an overall noise level of 160 decibels.

Author

Environmental Tests; Heat Shielding; Composite Materials; Aerodynamic Heating; Insulation; Heat Resistant Alloys; Thermal Resistance; Insulated Structures

19980228014 NASA Langley Research Center, Hampton, VA USA

Landing Characteristics of a Lenticular-Shaped Reentry Vehicle

Blanchard, Ulysse J., NASA Langley Research Center, USA; Sep. 1961; 32p; In English

Report No.(s): NASA-TN-D-940; L-1676; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An experimental investigation was made of the landing characteristics of a 1/9-scale dynamic model of a lenticular-shaped reentry vehicle having extendible tail panels for control after reentry and for landing control (flare-out). The landing tests were made by catapulting a free model onto a hard-surface runway and onto water. A "belly-landing" technique in which the vehicle was caused to skid and rock on its curved undersurface (heat shield), converting sinking speed into angular energy, was investigated on a hard-surface runway. Landings were made in calm water and in waves both with and without auxiliary landing devices. Landing motions and acceleration data were obtained over a range of landing attitudes and initial sinking speeds during hard-surface landings and for several wave conditions during water landings. A few vertical landings (parachute letdown) were made in calm water. The hard-surface landing characteristics were good. Maximum landing accelerations on a hard surface were 5g and 18 radians per sq second over a range of landing conditions. Horizontal landings on water resulted in large violent rebounds and some diving in waves. Extreme attitude changes during rebound at initial impact made the attitude of subsequent impact random. Maximum accelerations for water landings were approximately 21g and 145 radians per sq second in waves 7 feet high. Various auxiliary water-landing devices produced no practical improvement in behavior. Reduction of horizontal speed and positive control of impact attitude did improve performance in calm water. During vertical landings in calm water maximum accelerations of 15g and 110 radians per sq second were measured for a contact attitude of -45 deg and a vertical velocity of 70 feet per second.

Author

Aerodynamic Characteristics; Lenticular Bodies; Reentry Vehicles; Spacecraft Landing; Water Landing

19980228034 NASA Lewis Research Center, Cleveland, OH USA

Experimental Altitude Performance of JP-4 Fuel and Liquid-Oxygen Rocket Engine with an Area Ratio of 48

Fortini, Anthony, NASA Lewis Research Center, USA; Hendrix, Charles D., NASA Lewis Research Center, USA; Huff, Vearl N., NASA Lewis Research Center, USA; May 1959; 30p; In English

Report No.(s): NASA-MEMO-5-14-59E; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The performance for four altitudes (sea-level, 51,000, 65,000, and 70,000 ft) of a rocket engine having a nozzle area ratio of 48.39 and using JP-4 fuel and liquid oxygen as a propellant was evaluated experimentally by use of a 1000-pound-thrust engine operating at a chamber pressure of 600 pounds per square inch absolute. The altitude environment was obtained by a rocket-ejector system which utilized the rocket exhaust gases as the pumping fluid of the ejector. Also, an engine having a nozzle area ratio of 5.49 designed for sea level was tested at sea-level conditions. The following table lists values from faired experimental curves at an oxidant-fuel ratio of 2.3 for various approximate altitudes.

Author

JP-4 Jet Fuel; Altitude; Rocket Engines; Rocket Exhaust; Liquid Oxygen

19980228052 NASA Ames Research Center, Moffett Field, CA USA

Three-Dimensional Orbits of Earth Satellites, Including Effects of Earth Oblateness and Atmospheric Rotation

Nielsen, Jack N., NASA Ames Research Center, USA; Goodwin, Frederick K., NASA Ames Research Center, USA; Mersman, William A., NASA Ames Research Center, USA; Dec. 1958; 84p; In English

Report No.(s): NASA-MEMO-12-4-58A; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

The principal purpose of the present paper is to present sets of equations which may be used for calculating complete trajectories of earth satellites from outer space to the ground under the influence of air drag and gravity, including oblateness effects, and to apply these to several examples of entry trajectories starting from a circular orbit. Equations of motion, based on an "instantaneous ellipse" technique, with polar angle as independent variable, were found suitable for automatic computation of orbits in which the trajectory consists of a number of revolutions. This method is suitable as long as the trajectory does not become nearly vertical. In the terminal phase of the trajectories, which are nearly vertical, equations of motion in spherical polar coordinates with time as the independent variable were found to be more suitable. In the first illustrative example the effects of the oblateness component of the earth's gravitational field and of atmospheric rotation were studied for equatorial orbits. The satellites were launched

into circular orbits at a height of 120 miles, an altitude sufficiently high that a number of revolutions could be studied. The importance of the oblateness component of the earth's gravitational field is shown by the fact that a satellite launched at circular orbital speed, neglecting oblateness, has a perigee some 67,000 feet lower when oblateness forces are included in the equations of motion than when they are not included. Also, the loss in altitude per revolution is double that of a satellite following an orbit not subject to oblateness. The effect of atmospheric rotation on the loss of altitude per revolution was small. As might be surmised, the regression of the line of nodes as predicted by celestial mechanics is unchanged when drag is included. It is clear that the inclination of the orbital plane to the equator will be relatively unaffected by drag for no atmospheric rotation since the drag lies in the orbital plane in this case. With the inclusion of atmospheric rotation it was found that the inclination of the plane changed about one-millionth of a radian per revolution. Thus the prediction of the position of the orbital plane of an earth satellite is not complicated by the introduction of drag. The line of apsides, which without drag but with oblateness moves slowly in space, tends to move with the satellite when drag is included in the calculations. As a results, the usual linearized solutions based on oblateness alone must be basically altered when drag is included to take into account the rapid movement of the line of apsides. In the second illustrative example the final revolution was calculated to impact for a number of trajectories in an orbital plane inclined at 65° to the equator. Of particular interest is the large effect the oblateness gravitational field and atmospheric rotation can have on the impact point. For a value of CDA/m of unity, and for an initial downward angle at 80 miles altitude of 0.01 radian, such as might be utilized for manned re-entry, oblateness had an influence of about 300 miles in the impact point, and atmospheric rotation had about a 150-mile influence.

Author

Aerodynamic Drag; Spherical Coordinates; Gravitational Fields; Equatorial Orbits; Circular Orbits

19980228267 NASA Langley Research Center, Hampton, VA USA

Landing Energy Dissipation for Manned Reentry Vehicles

Fisher, Lloyd J., Jr., NASA Langley Research Center, USA; Sep. 1960; 18p; In English

Report No.(s): NASA-TN-D-453; L-1082; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Analytical and experimental investigations have been made to determine the landing-energy-dissipation characteristics for several types of landing gear for manned reentry vehicles. The landing vehicles are considered in two categories: those having essentially vertical-descent paths, the parachute-supported vehicles, and those having essentially horizontal paths, the lifting vehicles. The energy-dissipation devices discussed are crushable materials such as foamed plastics and honeycomb for internal application in couch-support systems, yielding metal elements as part of the structure of capsules or as alternates for oleos in landing-gear struts, inflatable bags, braking rockets, and shaped surfaces for water impact. It appears feasible to readily evaluate landing-gear systems for internal or external application in hard-surface or water landings by using computational procedures and free-body landing techniques with dynamic models. The systems investigated have shown very interesting energy-dissipation characteristics over a considerable range of landing parameters. Acceptable gear can be developed along lines similar to those presented if stroke requirements and human-tolerance limits are considered.

Author

Aerodynamic Characteristics; Landing Gear; Struts; Reentry Vehicles; Energy Dissipation; Manned Reentry

19980228348 NASA Langley Research Center, Hampton, VA USA

Low-Subsonic-Speed Static Longitudinal Stability and Control Characteristics of a Winged Reentry-Vehicle Configuration Having Wingtip Panels that Fold up for High-Drag Reentry

Ware, George M., NASA Langley Research Center, USA; Feb. 1960; 18p; In English

Report No.(s): NASA-TM-X-227; L-747; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation of the low-subsonic-speed static longitudinal stability and control characteristics of a model of a manned reentry-vehicle configuration capable of high-drag reentry and glide landing has been made in the Langley free-flight tunnel. The model had a modified 63 deg delta plan-form wing with a fuselage on the upper surface. This configuration had wingtip panels designed to fold up 90 deg for the high-drag reentry phase of the flight and to extend horizontally for the glide landing. Data for the basic configurations and modifications to determine the effects of plan form, wingtip panel incidence, dihedral, and vertical position of the wingtip panels are presented without analysis.

Author

Longitudinal Stability; Static Stability; Subsonic Speed; Longitudinal Control; Manned Reentry; Delta Wings; Spacecraft Configurations; Wing Tips; Free Flight

19980228462 NASA Ames Research Center, Moffett Field, CA USA

Effects of Flight Conditions at Booster Separation on Payload Weight in Orbit

Nelson, Richard D., NASA Ames Research Center, USA; Nov. 1961; 48p; In English

Report No.(s): NASA-TN-D-1069; A-521; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A study of the principal flight parameters at booster separation was conducted to find the effect of each on the weight of the payload boosted into an earth orbit along a zero drag gravity turn trajectory. The parameters considered include Mach number (3 to 9), flight-path angle (10 deg to 55 deg), altitude (90,000 and 350,000 ft), inert weight ratio (0.05 to 0.15), and thrust-weight ratio (1.5 to 2.5), with a specific impulse of 289 seconds. Both transfer ellipse and direct orbit trajectories were considered. With either trajectory method, payload weight was highest for low initial flight-path angles and high initial Mach numbers. Of course, high initial Mach numbers require greater energy expenditures from the booster. Changes in initial altitude had little effect on payload weight, and only small gains were evident when thrust-weight ratio was increased.

Author

Payloads; Booster Rocket Engines; Flight Conditions; Earth Orbits; Transfer Orbits; Thrust-Weight Ratio; Flight Characteristics

19980231024 Boeing North American, Inc., Reusable Space Systems, Downey, CA USA

Shuttle Liquid Fly Back Booster Configuration Options

Healy, T. J., Jr., Boeing North American, Inc., USA; 1998; 14p; In English, 16-17 Jul. 1998, Cleveland, OH, USA; Sponsored by NASA, USA

Contract(s)/Grant(s): NAS8-97272; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

This paper surveys the basic configuration options available to a Liquid Fly Back Booster (LFBB), integrated with the Space Shuttle system. The background of the development of the LFBB concept is given. The influence of the main booster engine (BME) installations and the Fly Back Engine (FBE) installation on the aerodynamic configurations are also discussed. Limits on the LFBB configuration design space imposed by the existing Shuttle flight and ground elements are also described. The objective of the paper is to put the constraints and design space for an LFBB in perspective. The object of the work is to define LFBB configurations that significantly improve safety, operability, reliability and performance of the Shuttle system and dramatically lower operations costs.

Author

Design Analysis; Aerodynamic Configurations; Booster Rocket Engines; Space Shuttle Boosters; Cost Reduction

19980231044 NASA Ames Research Center, Moffett Field, CA USA

Motion and Heating During Atmosphere Reentry of Space Vehicles

Wong, Thomas J., NASA Ames Research Center, USA; Goodwin, Glen, NASA Ames Research Center, USA; Slye, Robert E., NASA Ames Research Center, USA; Sep. 1960; 20p; In English

Report No.(s): NASA-TN-D-334; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The results of an analysis of the motion and heating during atmospheric reentry of manned space vehicles has shown the following: 1. Flight-corridor depths which allow reentry in a single pass decrease rapidly as the reentry speed increases if the maximum deceleration is limited to 10 g. 2. Use of aerodynamic lift can result in a three-to five fold increase in corridor depth over that available to a ballistic vehicle for the same deceleration limits. 3. Use of aerodynamic lift to widen these reentry corridors causes a heating penalty which becomes severe for values of the lift-drag ratio greater than unity for constant lift-drag entry. 4. In the region of most intense convective heating the inviscid flow is generally in chemical equilibrium but the boundary-layer flows are out of equilibrium. Heating rates for the nonequilibrium boundary layer are generally lower than for the corresponding equilibrium case. 5. Radiative heating from the hot gas trapped between the shock wave and the body stagnation region may be as severe as the convective heating and unfortunately occurs at approximately the same time in the flight.

Author

Aerodynamic Heating; Reentry Effects; Reentry Vehicles; Manned Spacecraft; Convective Heat Transfer; Atmospheric Entry

19980231061 NASA Langley Research Center, Hampton, VA USA

Trajectory Control for Vehicles Entering the Earth's Atmosphere at Small Flight Path Angles

Eggleston, John M., NASA Langley Research Center, USA; Feb. 1959; 68p; In English

Report No.(s): NASA-MEMO-1-19-59L; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Methods of controlling the trajectories of high-drag-low-lift vehicles entering the earth's atmosphere at angles of attack near 90 deg and at initial entry angles up to 3 deg are studied. The trajectories are calculated for vehicles whose angle of attack can be held constant at some specified value or can be perfectly controlled as a function of some measured quantity along the trajectory.

The results might be applied in the design of automatic control systems or in the design of instruments which will give the human pilot sufficient information to control his trajectory properly during an atmospheric entry. Trajectory data are compared on the basis of the deceleration, range, angle of attack, and, in some cases, the rate of descent. The aerodynamic heat-transfer rate and skin temperature of a vehicle with a simple heat-sink type of structure are calculated for trajectories made with several types of control functions. For the range of entry angles considered, it is found that the angle of attack can be controlled to restrict the deceleration down to an arbitrarily chosen level of 3g. All the control functions tried are successful in reducing the maximum deceleration to the desired level. However, in order to avoid a tendency for the deceleration to reach an initial peak decrease, and then reach a second peak, some anticipation is required in the control function so that the change in angle of attack will lead the change in deceleration. When the angle of attack is controlled in the aforementioned manner, the maximum rate of aerodynamic heat transfer to the skin is reduced, the maximum skin temperature of the vehicle is virtually unaffected, and the total heat absorbed is slightly increased. The increase in total heat can be minimized, however, by maintaining the maximum desired deceleration for as much of the trajectory as possible. From an initial angle of attack of 90 deg, the angle-of-attack requirements necessary to maintain constant values of deceleration (1g to 4g) and constant values of rate of descent (450 to 1,130 ft/sec) as long as it is aerodynamically practical are calculated and are found to be moderate in both magnitude and rate. Entry trajectories made with these types of control are presented and discussed.

Author

Aerodynamic Heat Transfer; Angle of Attack; Atmospheric Entry; Trajectory Control; Heat Sinks; Earth Atmosphere; Trajectories

11

CHEMISTRY AND MATERIALS

Includes chemistry and materials (general); composite materials; inorganic and physical chemistry; metallic materials; nonmetallic materials; propellants and fuels; and materials processing.

19980228308 NASA Ames Research Center, Moffett Field, CA USA

Experimental Investigation of Aerodynamic Effects of External Combustion in Airstream Below Two-Dimensional Supersonic Wing at Mach 2.5 and 3.0

Dorsch, Robert G., NASA Ames Research Center, USA; Serafini, John S., NASA Ames Research Center, USA; Fletcher, Edward A., NASA Ames Research Center, USA; Pinkel, I. Irving, NASA Ames Research Center, USA; Mar. 1959; 22p; In English Report No.(s): NASA-MEMO-1-11-59E; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Pressure distributions associated with stable combustion of aluminum borohydride in the airstream adjacent to the lower surface of a 13-inch chord, two-dimensional, blunt-base wing were determined experimentally. The measurements were made with the wing at 20 angle of attack in a 1- by 1-foot tunnel at Mach numbers of 2.47 and 2.96. Static-pressure increases along the lower surface and base caused by the combustion are presented along with the resultant lift increases. The lift-drag ratio of the wing was nearly doubled by the addition of heat. The experimental values of lift during heat addition agree with those predicted by analytical calculations.

Author

Aluminum Borohydrides; Pressure Distribution; Metal Combustion; Aerodynamics; Air Flow; Supersonic Speed; Wings; Wind Tunnel Tests

19980228309 NASA Lewis Research Center, Cleveland, OH USA

Performance of Two Boron-Modified S-816 Alloys in a Turbojet Engine Operated at 1650 F

Waters, William J., NASA Lewis Research Center, USA; Signorelli, Robert A., NASA Lewis Research Center, USA; Johnston, James R., NASA Lewis Research Center, USA; Mar. 1959; 32p; In English Report No.(s): NASA-MEMO-3-3-59E; E-229; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

S-816+B and modified S-816+B cast cobalt-base alloys were evaluated as turbine-bucket materials at 1650 F. Stress-rupture and tensile data obtained from these alloys had indicated satisfactory strength for engine operation at 1650 F. Although both alloys exhibited a limited ductility in room-temperature laboratory impact tests, there was a significant increase in impact resistance in the 1650 F tests. Bucket failures began after 10 hours of engine testing and continued at various intervals during the 107.5-hour test. Bucket lives were short relative to the predicted lives based on stress-rupture considerations (280 hr for S-816+B and 1750 hr for modified S-816+B). No significant difference was apparent in the performance of the two alloy groups. The primary cause of bucket failures in both alloys was mechanical fatigue. Impact damage occurred as a direct result of bucket tip fatigue failures and was a secondary cause of bucket failures. The impact of small pieces of fractured bucket tips on surrounding buckets caused

a relatively large amount of impact damage to buckets of both alloys. The amount of impact damage from induced fractures at the bucket midspan, which provided relatively large failed fragments, was no greater than that which occurred as a result of tip failures.

Author

Turbojet Engines; Aircraft Construction Materials; Boron Alloys; Cast Alloys; High Strength Alloys; Impact Tests; Impact Resistance; Mechanical Properties; Impact Damage

19980231059 NASA Lewis Research Center, Cleveland, OH USA

Exploratory Investigation of Advanced-Temperature Nickel-Base Alloys

Freche, John C., NASA Lewis Research Center, USA; Waters, William J., NASA Lewis Research Center, USA; May 1959; 42p; In English

Report No.(s): NASA-MEMO-4-13-59E; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation was conducted to provide an advanced-temperature nickel-base alloy with properties suitable for aircraft turbine blades as well as for possible space vehicle applications. An entire series of alloys that do not require vacuum melting techniques and that generally provide good stress-rupture and impact properties was evolved. The basic-alloy composition of 79 percent nickel, 8 percent molybdenum, 6 percent chromium, 6 percent aluminum, and 1 percent zirconium was modified by a series of element additions such as carbon, titanium, and boron, with the nickel content adjusted to account for the additives. Stress-rupture, impact, and swage tests were made with all the alloys. The strongest composition (basic alloy plus 1.5 percent titanium plus 0.125 percent carbon) displayed 384- and 574-hour stress-rupture lives at 1800 F and 15,000 psi in the as-cast and homogenized conditions, respectively. All the alloys investigated demonstrated good impact resistance. Several could not be broken in a low-capacity Izod impact tester and, on this basis, all compared favorably with several high-strength high-temperature alloys. Swaging cracks were encountered with all the alloys. In several cases, however, these cracks were slight and could be detected only by zygl examination. Some of these compositions may become amenable to hot working on further development. On the basis of the properties indicated, it appears that several of the alloys evolved, particularly the 1.5 percent titanium plus 0.125 percent carbon basic-alloy modification, could be used for advanced- temperature turbine blades, as well as for possible space vehicle applications.

Author

Research; Procedures; Nickel Alloys; Temperature Effects; Mechanical Properties; Technology Utilization; Gas Turbine Engines

19980231086 NASA, Washington, DC USA

Lubricating Oils for Aviation Gas Turbines

Panov, V. V.; Sobolev, Yu. S.; May 1980; 82p; In English; Translated into English by Translations, 130 West 57th Street, New York 19, NY

Report No.(s): NASA-TT-F-21; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

The ultimate serviceability of gas turbine engine lubricating oils is assessed on the basis of results of tests on individual frictional assembly installations of engines and on full-scale engines. In certain cases, oils are tested under in-flight conditions.

Author

Lubricating Oils; Gas Turbine Engines; Installing

12 ENGINEERING

Includes engineering (general); communications and radar; electronics and electrical engineering; fluid mechanics and heat transfer; instrumentation and photography; lasers and masers; mechanical engineering; quality assurance and reliability; and structural mechanics.

19980227962 NASA Langley Research Center, Hampton, VA USA

Investigation of the Flow Over Simple Bodies at Mach Numbers of the Order of 20

Henderson, Arthur, Jr., NASA Langley Research Center, USA; Aug. 1960; 22p; In English

Report No.(s): NASA-TN-D-449; L-1051; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

It is shown that adequate means are available for calculating inviscid direct and induced pressures on simple axisymmetric bodies at zero angle of attack. The extent to which viscous effects can alter these predictions is indicated. It is also shown that

inviscid induced pressures can significantly affect the stability of blunt, two-dimensional flat wings at low angles of attack. However, at high angles of attack, the inviscid induced pressure effects are negligible.

Author

Axisymmetric Bodies; Conical Bodies; Angle of Attack; Viscous Flow; Mach Number; Aerodynamic Characteristics; Aerodynamic Balance; Aerodynamic Configurations; Aeronautical Engineering

19980227969 NASA Ames Research Center, Moffett Field, CA USA

Two-Dimensional, Supersonic, Linearized Flow with Heat Addition

Lomax, Harvard, NASA Ames Research Center, USA; Feb. 1959; 38p; In English

Report No.(s): NASA-MEMO-1-10-59A; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Calculations are presented for the forces on a thin supersonic wing underneath which the air is heated. The analysis is limited principally to linearized theory but nonlinear effects are considered. It is shown that significant advantages to external heating would exist if the heat were added well below and ahead of the wing.

Author

Two Dimensional Flow; Supersonic Flow; Heating; Thin Wings; Propulsion

19980227973 NASA Ames Research Center, Moffett Field, CA USA

Effects of Mach Number, Leading-Edge Bluntness, and Sweep on Boundary-Layer Transition on a Flat Plate

Jillie, Don W., NASA Ames Research Center, USA; Hopkins, Edward J., NASA Ames Research Center, USA; Sep. 1961; 32p; In English

Report No.(s): NASA-TN-D-1071; A-481; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The effects of leading-edge bluntness and sweep on boundary-layer transition on flat plate models were investigated at Mach numbers of 2.00, 2.50, 3.00, and 4.00. The effect of sweep on transition was also determined on a flat plate model equipped with an elliptical nose at a Mach number of 0.27. Models used for the supersonic investigation had leading-edge radii varying from 0.0005 to 0.040 inch. The free-stream unit Reynolds number was held constant at 15 million per foot for the supersonic tests and the angle of attack was 0 deg. Surface flow conditions were determined by visual observation and recorded photographically. The sublimation technique was used to indicate transition, and the fluorescent-oil technique was used to indicate flow separation. Measured Mach number and sweep effects on transition are compared with those predicted from shock-loss considerations as described in NACA Rep. 1312. For the models with the blunter leading edges, the transition Reynolds number (based on free-stream flow conditions) was approximately doubled by an increase in Mach number from 2.50 to 4.00; and nearly the same result was predicted from shock-loss considerations. At all supersonic Mach numbers, increases in sweep reduced the transition Reynolds number and the amount of reduction increased with increases in bluntness. The shock-loss method considerably underestimated the sweep effects, possibly because of the existence of crossflow instability associated with swept wings. At a Mach number of 0.27, no reduction in the transition Reynolds number with sweep was measured (as would be expected with no shock loss) until the sweep angle was attained where crossflow instability appeared.

Author

Boundary Layer Transition; Mach Number; Blunt Bodies; Sweep Effect; Supersonic Speed; Boundary Layer Separation; Swept Wings; Free Flow; Flat Plates; Cross Flow

19980227976 NASA Ames Research Center, Moffett Field, CA USA

Some Effects of Leading-Edge Sweep on Boundary-Layer Transition at Supersonic Speeds

Chapman, Gray T., NASA Ames Research Center, USA; Sep. 1961; 38p; In English

Report No.(s): NASA-TN-D-1075; A-461; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The effects of crossflow and shock strength on transition of the laminar boundary layer behind a swept leading edge have been investigated analytically and with the aid of available experimental data. An approximate method of determining the crossflow Reynolds number on a leading edge of circular cross section at supersonic speeds is presented. The applicability of the critical crossflow criterion described by Owen and Randall for transition on swept wings in subsonic flow was examined for the case of supersonic flow over swept circular cylinders. A wide range of applicability of the subsonic critical values is indicated. The corresponding magnitude of crossflow velocity necessary to cause instability on the surface of a swept wing at supersonic speeds was also calculated and found to be small. The effects of shock strength on transition caused by Tollmien-Schlichting type of instability are discussed briefly. Changes in local Reynolds number, due to shock strength, were found analytically to have considerably more effect on transition caused by Tollmien-Schlichting instability than on transition caused by crossflow instability. Changes in the

mechanism controlling transition from Tollmien-Schlichting instability to crossflow instability were found to be possible as a wing is swept back and to result in large reductions in the length of laminar flow.

Author

Supersonic Speed; Leading Edge Sweep; Laminar Boundary Layer; Boundary Layer Transition; Cross Flow; Circular Cylinders; Swept Wings

19980228001 NASA Langley Research Center, Hampton, VA USA

Heat-Transfer and Pressure Measurements from a Flight Test of the Third 1/18-Scale Model of the Titan Intercontinental Ballistic Missile up to a Mach Number of 3.86 and Reynolds Number per Foot of 23.5×10^6 and a Comparison with Heat Transfer

Graham, John B., Jr., NASA Langley Research Center, USA; Oct. 1958; 62p; In English

Report No.(s): NASA-MEMO-11-1-58L; AF-AM-70; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

Heat-transfer and pressure measurements were obtained from a flight test of a 1/18-scale model of the Titan intercontinental ballistic missile up to a Mach number of 3.86 and Reynolds number per foot of 23.5×10^6 and are compared with the data of two previously tested 1/18-scale models. Boundary-layer transition was observed on the nose of the model. Van Driest's theory predicted heat-transfer coefficients reasonably well for the fully laminar flow but predictions made by Van Driest's theory for turbulent flow were considerably higher than the measurements when the skin was being heated. Comparison with the flight test of two similar models shows fair repeatability of the measurements for fully laminar or turbulent flow.

Author

Intercontinental Ballistic Missiles; Heat Transfer; Pressure Measurement; Turbulent Flow; Flight Tests; Titan; Scale Models; Laminar Flow; Aerodynamic Heating

19980228005 NASA Langley Research Center, Hampton, VA USA

Measurements of the Time-Averaged and Instantaneous Induced Velocities in the Wake of a Helicopter Rotor Hovering at High Tip Speeds

Heyson, Harry H., NASA Langley Research Center, USA; Jul. 1960; 46p; In English

Report No.(s): NASA-TN-D-393; L-836; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Measurements of the time-averaged induced velocities were obtained for rotor tip speeds as great as 1,100 feet per second (tip Mach number of 0.98) and measurements of the instantaneous induced velocities were obtained for rotor tip speeds as great as 900 feet per second. The results indicate that the small effects on the wake with increasing Mach number are primarily due to the changes in rotor-load distribution resulting from changes in Mach number rather than to compressibility effects on the wake itself. No effect of tip Mach number on the instantaneous velocities was observed. Under conditions for which the blade tip was operated at negative pitch angles, an erratic circulatory flow was observed.

Author

Rotary Wings; Blade Tips; Helicopters; Tip Speed; Rotors; Helicopter Performance

19980228013 NASA Lewis Research Center, Cleveland, OH USA

Measurement of Screen-Size Effects on Intensity, Scale, and Spectrum of Turbulence in a Free Subsonic Jet

Howard, Charles D., NASA Lewis Research Center, USA; Laurence, James C., NASA Lewis Research Center, USA; Aug. 1960; 38p; In English

Report No.(s): NASA-TN-D-297; E-798; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The results are reported of hot-wire anemometer measurements of the fluctuating longitudinal component of the turbulent velocities in the mean flow downstream of screens in an air jet. These measurements have been analyzed by well-established techniques to give the influence of tile screen mesh size on the turbulent intensity, scale, and the power-spectral-density. The results show a linear dependence of the intensity upon the screen mesh size for locations within the central core of the air jet. The spectral-density curves show that the screens redistribute the turbulent energy from the low frequencies (<1000 cps) to the high frequencies (>1000 cps). The effects of the screens are overwhelmed in the mixing region of the jet flow by the turbulence levels existing there. The large pressure drops occurring across the screens reduce the velocity of the jet as compared to the jet without screens by approximately one-third for the velocity and range of mesh sizes investigated and reported in this report. The turbulence scale is a linear function of distance from the nozzle exit and is somewhat greater than comparable jets without screens.

Author

Hot-Wire Anemometers; Turbulence; Subsonic Flow; Jet Flow; Free Jets; Aerodynamic Noise; Velocity Measurement; Flow Velocity

19980228029 NASA Lewis Research Center, Cleveland, OH USA

Evaluation of Transpiration-Cooled Turbine Blades with Shells of "Poroloy" Wire Cloth

Richards, Hadley T., NASA Lewis Research Center, USA; May 1959; 20p; In English

Report No.(s): NASA-MEMO-1-29-59E; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An experimental investigation was made to evaluate the durability and permeability of a group of transpiration-cooled, strut-supported turbine blades. The porous shells were formed from a woven-wire material. The blades were fabricated by a contractor for the Bureau of Aeronautics. The results of permeability tests indicated that the shell material exhibited large random variations in local permeability, which result in excessive coolant flows and very nonuniform cooling. For this reason no heat-transfer evaluations were made because any results would have been inconclusive. Four blades were investigated for structural soundness in a turbo-jet engine operating at a turbine-inlet temperature of approximately 1670 deg F and a turbine tip speed of approximately 1305 feet per second. The maximum temperature of the porous-shell material was approximately 1050 deg F. Inspection of the first two blades after 10 minutes of engine operation revealed that the tips of both of the blades had failed. For the second pair of blades, an improved tip cap was provided by the use of built-up weld extending from strut tip to shell. One of these blades was then operated for 33 hours without failure, and was found to be in good condition at the end of this time. The second blade of this second pair failed within the first 10 minutes of operation because of a poor bond between shell and strut lands.

Author

Transpiration; Turbine Blades; Cooling; Heat Transfer; Jet Engines; Wire Cloth

19980228040 NASA Langley Research Center, Hampton, VA USA

Water-Landing Characteristics of a Reentry Capsule

McGehee, John R., NASA Langley Research Center, USA; Hathaway, Melvin E., NASA Langley Research Center, USA; Vaughan, Victor L., Jr., NASA Langley Research Center, USA; Jun. 1959; 28p; In English

Report No.(s): NASA-MEMO-5-23-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Experimental and theoretical investigations have been made to determine the water-landing characteristics of a conical-shaped reentry capsule having a segment of a sphere as the bottom. For the experimental portion of the investigation, a 1/12-scale model capsule and a full-scale capsule were tested for nominal flight paths of 65 deg and 90 deg (vertical), a range of contact attitudes from -30 deg to 30 deg, and a full-scale vertical velocity of 30 feet per second at contact. Accelerations were measured by accelerometers installed at the centers of gravity of the model and full-scale capsules. For the model test the accelerations were measured along the X-axis (roll) and Z-axis (yaw) and for the full-scale test they were measured along the X-axis (roll), Y-axis (pitch), and Z-axis (yaw). Motions and displacements of the capsules that occurred after contact were determined from high-speed motion pictures. The theoretical investigation was conducted to determine the accelerations that might occur along the X-axis when the capsule contacted the water from a 90 deg flight path at a 0 deg attitude. Assuming a rigid body, computations were made from equations obtained by utilizing the principle of the conservation of momentum. The agreement among data obtained from the model test, the full-scale test, and the theory was very good. The accelerations along the X-axis, for a vertical flight path and 0 deg attitude, were in the order of 40g. For a 65 deg flight path and 0 deg attitude, the accelerations along the X-axis were in the order of 50g. Changes in contact attitude, in either the positive or negative direction from 0 deg attitude, considerably reduced the magnitude of the accelerations measured along the X-axis. Accelerations measured along the Y- and Z-axes were relatively small at all test conditions.

Author

Water Landing; Numerical Analysis; Experimentation; Data Acquisition; Conical Nozzles; Full Scale Tests; Aerodynamic Characteristics; Reentry Vehicles

19980228053 NASA Langley Research Center, Hampton, VA USA

Effects of Cross-Sectional Shape, Solidity, and Distribution of Heat-Transfer Coefficient on the Torsional Stiffness of Thin Wings Subjected to Aerodynamic Heating

Thomson, Robert G., NASA Langley Research Center, USA; Feb. 1959; 34p; In English

Report No.(s): NASA-MEMO-1-30-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A study has been made of the effects of varying the shape, solidity, and heat-transfer coefficient of thin wings with regard to their influence on the torsional-stiffness reduction induced by aerodynamic heating. The variations in airfoil shape include blunting, flattening, and combined blunting and flattening of a solid wing of symmetrical double-wedge cross section. Hollow double-wedge wings of constant skin thickness with and without internal webs also are considered. The effects of heat-transfer coefficients appropriate for laminar and turbulent flow are investigated in addition to a step transition along the chord from a lower to a higher constant value of heat-transfer coefficient. From the results given it is concluded that the flattening of a solid double wedge decreases the reduction in torsional stiffness while slight degrees of blunting increase the loss. The influence of chordwise

variations in heat-transfer coefficient due to turbulent and laminar boundary-layer flow on the torsional stiffness of solid wings is negligible. The effect of a step transition in heat-transfer coefficient along the chord of a solid wing can, however, become appreciable. The torsional-stiffness reduction of multiweb and hollow double-wedge wings is substantially less than that calculated for a solid wing subjected to the same heating conditions.

Author

Heat Transfer Coefficients; Stiffness; Torsion; Thin Wings; Aerodynamic Heating; Research; Collision Parameters

19980228054 NASA Langley Research Center, Hampton, VA USA

Hydrodynamic Characteristics of Two Low-Drag Supercavitating Hydrofoils

McGehee, John R., NASA Langley Research Center, USA; Johnson, Virgil E., Jr., NASA Langley Research Center, USA; Jun. 1959; 66p; In English

Report No.(s): NASA-MEMO-5-9-59L; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An experimental investigation has been conducted in Langley tank no. 2 to determine the hydrodynamic characteristics of two low-drag supercavitating hydrofoils operating in a range of cavitation numbers from 0 to approximately 6. The hydrofoils had aspect ratios of 1 and 3, and the sections were derived by assuming five terms in the vorticity-distribution expansion of the equivalent airfoil. The aspect-ratio-1 hydrofoil was also tested at zero cavitation number with two sets of end plates having depths of 3/8 and 1/4 chords. Zero cavitation number was established by operating the hydrofoils near the water surface so that complete ventilation of the upper surfaces could be obtained. For those depths of submersion where complete ventilation was not obtained through vortex ventilation, two probes were used to introduce air to the upper surfaces of the hydrofoils and to induce complete ventilation. Data were obtained for a range of speeds from 20 to 80 fps, angles of attack from 2 to 20 deg, and ratios of depth of submersion to chord from 0 to 0.85. The experimental results obtained from the aspect-ratio-1 and aspect-ratio-3, five-term hydrofoils were compared with a three-dimensional zero-cavitation-number theory. The theoretical and experimental values of lift and center of pressure for the aspect-ratio-1 hydrofoil were in agreement, within engineering accuracy, for the range of lift coefficients investigated. The theoretical drag coefficients were lower, by a constant amount, than the experimental drag coefficients. The theoretical expressions derived for the lift, drag, and center of pressure of the aspect-ratio-3 hydrofoil were in agreement, within engineering accuracy, with the experimental values. The theoretical and experimental drag coefficients of the aspect-ratio-3 five-term hydrofoil were lower than the theoretical drag coefficients computed for a comparable Tulin-Burkart hydrofoil.

Author

Supercavitating Flow; Hydrofoils; Aerodynamic Coefficients; Ventilation; Vortices; Hydrodynamics

19980228067 NASA Langley Research Center, Hampton, VA USA

Experimental and Theoretical Studies of Axisymmetric Free Jets

Love, Eugene S., NASA Langley Research Center, USA; Grigsby, Carl E., NASA Langley Research Center, USA; Lee, Louise P., NASA Langley Research Center, USA; Woodling, Mildred J., NASA Langley Research Center, USA; 1959; 298p; In English
Report No.(s): NASA-TR-R-6; No Copyright; Avail: CASI; A13, Hardcopy; A03, Microfiche

Some experimental and theoretical studies have been made of axisymmetric free jets exhausting from sonic and supersonic nozzles into still air and into supersonic streams with a view toward problems associated with propulsive jets and the investigation of these problems. For jets exhausting into still air, consideration is given to the effects of jet Mach number, nozzle divergence angle, and jet static pressure ratio upon jet structure, jet wavelength, and the shape and curvature of the jet boundary. Studies of the effects of the ratio of specific heats of the jets are included are observations pertaining to jet noise and jet simulation. For jets exhausting into supersonic streams, an attempt has been made to present primarily theoretical certain jet interference effects and in formulating experimental studies. The primary variables considered are jet Mach number, free stream Mach number, jet static pressure ratio, ratio of specific heats of the jet, nozzle exit angle, and boattail angle. The simulation problem and the case of a hypothetical hypersonic vehicle are examined. A few experimental observations are included.

Author

Free Jets; Supersonic Nozzles; Acoustic Simulation; Aerodynamic Interference; Supersonic Flow

19980228193 Georgia Inst. of Tech., Atlanta, GA USA

Experimental Smoke and Electromagnetic Analog Study of Induced Flow Field About a Model Rotor in Steady Flight Within Ground Effect

Gray, Robin B., Georgia Inst. of Tech., USA; Aug. 1960; 32p; In English

Report No.(s): NASA-TN-D-458; W-143; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Hovering and steady low-speed forward-flight tests were run on a 4-foot-diameter rotor at a ground height of 1 rotor radius. The two blades had a 2 to 1 taper ratio and were mounted in a see-saw hub. The solidity ratio was 0.05. Measurements were made

of the rotor rpm, collective pitch, and forward-flight velocity. Smoke was introduced into the tip vortex and the resulting vortex pattern was photographed from two positions. Using the data obtained from these photographs, wire models of the tip vortex configurations were constructed and the distribution of the normal component of induced velocity at the blade feathering axis that is associated with these tip vortex configurations was experimentally determined at 450 increments in azimuth position from this electromagnetic analog. Three steady-state conditions were analyzed. The first was hovering flight; the second, a flight velocity just under the wake "tuck under" speed; and the third, a flight velocity just above this speed. These corresponded to advance ratios of 0, 0.022, and 0.030 (or ratios of forward velocity to calculated hovering induced velocity of approximately 0, 0.48, and 0.65), respectively, for the model test rotor. Cross sections of the wake at 450 intervals in azimuth angle as determined from the path of the tip vortex are presented graphically for all three cases. The nondimensional normal component of the induced velocity that is associated with the tip vortex as determined by an electromagnetic analog at 450 increments in azimuth position and at the blade feathering axis is presented graphically. It is shown that the mean value of this component of the induced velocity is appreciably less after tuck-under than before. It is concluded that this method yields results of engineering accuracy and is a very useful means of studying vortex fields.

Derived from text

Experimentation; Smoke; Analog Data; Flight Tests; Hovering; Wakes; Flow Distribution

19980228215 NASA Langley Research Center, Hampton, VA USA

The Effect of Beam Loading on Water Impact Loads and Motions

Mixson, John S., NASA Langley Research Center, USA; Feb. 1959; 42p; In English

Report No.(s): NASA-MEMO-1-5-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation of the effect of beam loading on impact loads and motions has been conducted in the Langley impact basin. Water impact tests of flat-bottom 5-inch-and 8-inch-beam models having beam-loading coefficients $C(\text{sub } \Delta)$ from 62.5 to 544 and a 30 deg dead-rise 5-inch-beam model A having beam-loading coefficients from 208 to 530 are described and the results analyzed to show trends of these heavy-beam-loading data with initial flight-path angle, trim angle, dead-rise angle, and time through- out the impact. Data from flat-bottom model tests, $C(\text{sub } \Delta) = 4.4$ to 36.5, and from 30 deg dead-rise model tests, $C(\text{sub } \Delta) = 0.58$ and 18.8, are included, along with the heavy-beam-loading data; and variations of these data with beam-loading coefficients are shown. Each of the load and motion coefficients is found to be directly proportional to a power factor of $C(\text{sub } \Delta)$. For instance, the maximum impact lift coefficient is found $C(\text{sub } L, \text{max})$ to be directly proportional to $C(\text{sub } \Delta, \text{exp } 0.33)$ for the flat-bottom model and $C(\text{sub } \Delta, \text{exp } 0.45)$ for the 30 deg dead-rise model. These variations of $C(\text{sub } L, \text{max})$ with $C(\text{sub } \Delta)$ are found to be in agreement with theoretical variations. Finally, an empirical equation for the prediction of $C(\text{sub } L, \text{max})$ is presented and is shown to give good agreement with experimental $C(\text{sub } L, \text{max})$ for about 500 fixed-trim smooth-water impacts. The range of variables included dead-rise angles from 0 deg to deg, beam-loading coefficients from 0.48 to 544, trim angles from 3 deg to deg, and initial flight-path angles from about 2 deg to about deg.

Author

Impact Loads; Impact Tests; Aerodynamic Coefficients; Experimentation

19980228226 NASA Ames Research Center, Moffett Field, CA USA

Pressure Distributions at Transonic Speeds for Bumpy and Indented Midsections of a Basic Parabolic-Arc Body

Taylor, Robert A., NASA Ames Research Center, USA; Feb. 1959; 70p; In English

Report No.(s): NASA-MEMO-1-22-59A; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

The measured static-pressure distributions at the model surface and in the surrounding flow field are presented for a basic parabolic-arc body having a fineness ratio of 14 and for three additional bodies obtained by modifying the basic parabolic-arc body along the middle portion of the body length by adding a bump, by indenting, or by quadripole shaping. The data were obtained with the various bodies at zero angle of attack. The Mach number varied from 0.80 to 1.20 with a corresponding Reynolds number (based on body length) variation of $27 \times 10(\text{exp } 6)$ to $38 \times 10(\text{exp } 6)$. The data are subject to tunnel-wall interference and do not represent free-air conditions.

Author

Pressure Distribution; Transonic Speed; Flow Distribution; Aerodynamic Interference

19980228265 NASA Ames Research Center, Moffett Field, CA USA

Transition of the Laminar Boundary Layer on a Delta Wing with 74 degree Sweep in Free Flight at Mach Numbers from 2.8 to 5.3

Chapman, Gary T., NASA Ames Research Center, USA; Aug. 1961; 46p; In English

Report No.(s): NASA-TN-D-1066; A-589; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The tests were conducted at Mach numbers from 2.8 to 5.3, with model surface temperatures small compared to boundary-layer recovery temperature. The effects of Mach number, temperature ratio, unit Reynolds number, leading-edge diameter, and angle of attack were investigated in an exploratory fashion. The effect of heat-transfer condition (i.e., wall temperature to total temperature ratio) and Mach number can not be separated explicitly in free-flight tests. However, the data of the present report, as well as those of NACA TN 3473, were found to be more consistent when plotted versus temperature ratio. Decreasing temperature ratio increased the transition Reynolds number. The effect of unit Reynolds number was small as was the effect of leading-edge diameter within the range tested. At small values of angle of attack, transition moved forward on the windward surface and rearward on the leeward surface. This trend was reversed at high angles of attack (6 deg to 18 deg). Possible reasons for this are the reduction of crossflow on the windward side and the influence of the lifting vortices on the leeward surface. When the transition results on the 740 delta wing were compared to data at similar test conditions for an unswept leading edge, the results bore out the results of earlier research at nearly zero heat transfer; namely, sweep causes a large reduction in the transition Reynolds number.

Author

Laminar Boundary Layer; Delta Wings; Free Flight; Cross Flow; Heat Transfer; Temperature Ratio; Temperature Effects

19980228298 NASA Langley Research Center, Hampton, VA USA

Tire-to-Surface Friction Especially Under Wet Conditions

Sawyer, Richard H., NASA Langley Research Center, USA; Batterson, Sidney A., NASA Langley Research Center, USA; Harrin, Eziaslav N., NASA Langley Research Center, USA; Mar. 1959; 16p; In English

Report No.(s): NASA-MEMO-2-23-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The results of measurements of the maximum friction available in braking on various runway surfaces under various conditions is shown for a C-123B airplane and comparisons of measurements with a tire-friction cart on the same runways are made. The results of studies of wet-surface friction made with a 12-inch-diameter low-pressure tire on a tire-friction treadmill, with an automobile tire on the tire-friction cart, and with a 44 x 13 extra-high-pressure type VII aircraft tire at the Langley landing-load track are compared. Preliminary results of tests on the tire-friction treadmill under wet-surface conditions to determine the effect of the wiping action of the front wheel of a tandem-wheel arrangement on the friction available in braking for the rear wheel are given.

Author

Aircraft Tires; Friction; Sliding Friction; Surface Properties; Runways; Aircraft Brakes; Braking

19980228306 NASA Langley Research Center, Hampton, VA USA

Formulas Pertinent to the Calculation of Flow-Field Effects at Supersonic Speeds Due to Wing Thickness

Margolis, Kenneth, NASA Langley Research Center, USA; Elliott, Miriam H., NASA Langley Research Center, USA; May 1959; 26p; In English

Report No.(s): NASA-MEMO-4-3-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Expressions based on linearized supersonic-flow theory are derived for the perturbation velocity potential in space due to wing thickness for rectangular wings with biconvex airfoil sections and for arrow, delta, and quadrilateral wings with wedge-type airfoil sections. The complete range of supersonic speeds is considered subject to a minor aspect-ratio-Mach number restriction for the rectangular plan form and to the condition that the trailing edge is supersonic for the sweptback wings. The formulas presented can be utilized in determining the induced-flow characteristics at any point in the field and are readily adaptable for either numerical computation or analytical determination of any velocity components desired.

Author

Flow Characteristics; Supersonic Speed; Thickness; Airfoil Profiles; Delta Wings; Sweptback Wings; Arrow Wings; Rectangular Wings; Load Distribution (Forces)

19980228352 NASA Ames Research Center, Moffett Field, CA USA

Photographic Evidence of Streamwise Arrays of Vortices in Boundary-Layer Flow

Hopkins, Edward J., NASA Ames Research Center, USA; Keating, Stephen J., Jr., NASA Ames Research Center, USA; Bandetini, Angelo, NASA Ames Research Center, USA; Sep. 1960; 22p; In English

Report No.(s): NASA-TN-D-328; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Photographs are presented of various models coated with fluorescent oil to show evidence of surface vortices at a Mach number of 3.03. Vortex formation was evidently present on models with forward-facing steps, rearward-facing steps, wires, discrete surface particles, or unswept flat surfaces with sharp leading edges. Some photographs are also presented for the models coated with a sublimation material which clearly indicates the location of boundary-layer transition; however, it does not show the vortices as clearly as the fluorescent oil. The study was made on the models at an angle of attack of 0 deg at unit Reynolds numbers

of 7.7 and 10.7 million per foot. The spacing of the vortices as indicated by the flow studies on the unswept model was smaller at the higher Reynolds number in accordance with Gortler's theory. The flow studies also indicated that stable surface vortices produced by either steps or surface roughness persisted over model areas known to have turbulent boundary layers.

Author

Boundary Layer Transition; Photographs; Vortices; Turbulent Boundary Layer; Aerodynamic Configurations; Boundary Layer Flow; Supersonic Speed

19980228363 NASA Langley Research Center, Hampton, VA USA

Heat-Transfer Measurements on a 5.5- Inch-Diameter Hemispherical Concave Nose in Free Flight at Mach Numbers up to 6.6

Levine, Jack, NASA Langley Research Center, USA; Rumsey, Charles B., NASA Langley Research Center, USA; Dec. 1958; 48p; In English

Report No.(s): NASA-MEMO-10-21-58L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The aerodynamic heat transfer to a hemispherical concave nose has been measured in free flight at Mach numbers from 3.5 to 6.6 with corresponding Reynolds numbers based on nose diameter from 7.4×10^6 to 14×10^6 . Over the test Mach number range the heating on the cup nose, expressed as a ratio to the theoretical stagnation-point heating on a hemisphere nose of the same diameter, varied from 0.05 to 0.13 at the stagnation point of the cup, was approximately 0.1 at other locations within 40 deg of the stagnation point, and varied from 0.6 to 0.8 just inside the lip where the highest heating rates occurred. At a Mach number of 5 the total heat input integrated over the surface of the cup nose including the lip was 0.55 times the theoretical value for a hemisphere nose with laminar boundary layer and 0.76 times that for a flat face. The heating at the stagnation point was approximately 1/5 as great as steady-flow tunnel results. Extremely high heating rates at the stagnation point (on the order of 30 times the stagnation-point values of the present test), which have occurred in conjunction with unsteady oscillatory flow around cup noses in wind-tunnel tests at Mach and Reynolds numbers within the present test range, were not observed.

Author

Aerodynamic Heat Transfer; Concavity; Supersonic Speed; Unsteady Flow; Wind Tunnel Tests; Steady Flow; Hypersonic Speed; Free Flight; Aerodynamic Heating; Noses (Forebodies)

19980228364 NASA Langley Research Center, Hampton, VA USA

Flight Investigation of the Surface Pressure Distribution and Flow Field Around an Elliptical Spinner

Thomas, Lovic P., III, NASA Langley Research Center, USA; Feb. 1959; 12p; In English

Report No.(s): NASA-MEMO-1-26-59L; L-150; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

A flight investigation has been made of the surface pressure distribution and the flow field around a dummy, nonrotating, elliptical spinner over a Mach number range from 0.65 to 0.95, which corresponds to a Reynolds number range from about 1.6×10^6 per foot to about 3.9×10^6 per foot. The results showed that free-stream conditions were approximated from about 15 to 90 percent of the spinner length, but the local Mach number in the propeller plane varied from about 5 percent less than free stream at a Mach number of 0.65 to about 10 percent less than free stream at a mach number of 0.95.

Author

Pressure Distribution; Flow Distribution; Free Flow; Subsonic Speed; Flight Tests; Propellers; Spin Stabilization

19980228370 NASA Ames Research Center, Moffett Field, CA USA

Boundary-Layer Transition on Hollow Cylinders in Supersonic Free Flight as Affected by Mach Number and a Screw-thread Type of Surface Roughness

James, Carlton S., NASA Ames Research Center, USA; Feb. 1959; 54p; In English

Report No.(s): NASA-MEMO-1-20-59A; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

The effects of Mach number and surface-roughness variation on boundary-layer transition were studied using fin-stabilized hollow-tube models in free flight. The tests were conducted over the Mach number range from 2.8 to 7 at a nominally constant unit Reynolds number of 3 million per inch, and with heat transfer to the model surface. A screwthread type of distributed two-dimensional roughness was used. Nominal thread heights varied from 100 microinches to 2100 microinches. Transition Reynolds number was found to increase with increasing Mach number at a rate depending simultaneously on Mach number and roughness height. The laminar boundary layer was found to tolerate increasing amounts of roughness as Mach number increased. For a given

Mach number an optimum roughness height was found which gave a maximum laminar run greater than was obtained with a smooth surface.

Author

Boundary Layer Transition; Supersonic Flight; Supersonic Speed; Surface Roughness; Laminar Boundary Layer; Free Flight; Fins

19980228394 NASA Langley Research Center, Hampton, VA USA

Preliminary Heat-Transfer Measurements on a Hypersonic Glide Configuration Having 79.5 degree Sweepback and 45 degree Dihedral at a Mach Number of 4.95

Stainback, Calvin, NASA Langley Research Center, USA; Feb. 1960; 46p; In English

Report No.(s): NASA-TM-X-247; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An experimental investigation was conducted to evaluate the heat-transfer characteristics of a hypersonic glide configuration having 79.5 deg of sweepback (measured in the plane of the leading edges) and 45 of dihedral. The tests were conducted at a nominal Mach number of 4.95 and a stagnation temperature of 400 F. The test-section unit Reynolds number was varied from 1.95×10^6 to 12.24×10^6 per foot. The results indicated that the laminar-flow heat-transfer rate to the lower surface of the model decreased as the distance from the ridge line increased except for thermocouples located near the semispan at an angle of attack of 00 with respect to the plane of the leading edges. The heat-transfer distribution (local heating rate relative to the ridge-line heating rate) was similar to the theoretical heat-transfer distribution for a two-dimensional blunt body, if the ridge line was assumed to be the stagnation line, and could be predicted by this theory provided a modified Newtonian pressure distribution was used. Except in the vicinity of the apex, the ridge-line heat-transfer rate could also be predicted from two-dimensional blunt-body heat-transfer theory provided it was assumed that the stagnation-line heat-transfer rate varied as the cosine of the effective sweep (sine of the angle of attack of the ridge line). The heat-transfer level on the lower surface and the nondimensional heat-transfer distribution around the body on the lower surface were in qualitative agreement with the results of a geometric study of highly swept delta wings with large positive dihedrals made in reference 1.

Author

Heat Transfer; Gliding; Hypersonics; Laminar Flow; Dihedral Angle; Angle of Attack; Pressure Distribution; Swept Wings

19980228447 NASA Langley Research Center, Hampton, VA USA

Jet-Boundary Corrections for Lifting Rotors Centered in Rectangular Wind Tunnels

Heyson, Harry H., NASA Langley Research Center, USA; 1960; 66p; In English

Report No.(s): NASA-TR-R-71; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

A theory is developed and numerical corrections are computed. At high speeds, the corrections are the same as those for a wing. At low speeds, there is a large upwash at the rotor. Considerable care is required in the application of the results to very low speed flight conditions.

Author

Lifting Rotors; Jet Boundaries; Flight Conditions

19980228461 NASA Langley Research Center, Hampton, VA USA

Experimental Investigation of Two Low-Drag Supercavitating Hydrofoils at Speeds up to 200 Feet per Second

Christopher, Kenneth W., NASA Langley Research Center, USA; Johnson, Virgil E., Jr., NASA Langley Research Center, USA; Aug. 1960; 32p; In English

Report No.(s): NASA-TN-D-436; L-913; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An experimental investigation has been made in the Langley highspeed hydrodynamics facility to determine the force and moment characteristics of two hydrofoils (one having an aspect ratio of 1 and the other having an aspect ratio of 3) designed to have improved lift-drag ratios when operating under either supercavitating or ventilated conditions. Measurements were made of lift, drag, and pitching moment over a range of angles of attack from 40 to 200 for depths of submersion varying from 0 to approximately 1 chord. The range of speed for the investigation was from 110 to 200 feet per second. When the upper surface of the hydrofoils was completely unwetted, the experimental values of lift and drag forces were in good agreement with the theoretical values obtained from the zero-cavitation-number theory. The theoretical values for minimum angle of attack for operation with the upper surface of the hydrofoil unwetted define the lower limits of angle of attack for which the experimental values of lift coefficient are either in agreement with or slightly greater than those predicted by theory.

Author

Aerodynamic Coefficients; Hydrodynamics; Pitching Moments; Cavitation Flow; Lift Drag Ratio; Hydrofoils

19980228463 NASA, Washington, DC USA

The Laminar Boundary Layer at Hypersonic Speeds *Sullo Strato Limite Laminare in Corrente Ipersonica*

Ferrari, Carlo; Aerotecnica; Sep. 1961; Volume 36, No. 2, pp. 68-94; In English

Report No.(s): NASA-TT-F-71; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

A theoretical study of the laminar boundary layer on an airfoil immersed in a hypersonic stream is made under the assumption that (a) there is no heat transfer to the wall, and (b) there is zero gradient of pressure normal to the direction of development of the layer along the wall. Numerical applications are presented to illustrate the use of the theory for flow along a flat plate and for flow along a curved wall for which the shape is specified by certain governing profile parameters.

Author

Hypersonic Speed; Laminar Boundary Layer; Flat Plates; Airfoils; Heat Transfer; Hypersonic Flow

19980230613 NASA Langley Research Center, Hampton, VA USA

Experimental Influence Coefficients and Vibration Modes

Weidman, Deene J., NASA Langley Research Center, USA; Kordes, Eldon E., NASA Langley Research Center, USA; May 1959; 32p; In English

Report No.(s): NASA-MEMO-2-4-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Test results are presented for both symmetrical and antisymmetrical static loading of a wing model mounted on a three-point support system. The first six free-free vibration modes were determined experimentally. A comparison is made of the symmetrical nodal patterns and frequencies with the symmetrical nodal patterns and frequencies calculated from the experimental influence coefficients.

Author

Wings; Structural Analysis; Free Vibration; Structural Vibration; Wing Loading

19980231019 NASA Langley Research Center, Hampton, VA USA

Similar Solutions for the Compressible Boundary Layer on a Yawed Cylinder with Transpiration Cooling

Beckwith, Ivan E., NASA Langley Research Center, USA; 1959; 40p; In English

Report No.(s): NASA-TR-R-42; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Heat-transfer and skin-friction parameters obtained from exact solutions to the laminar compressible boundary-layer equations for infinite cylinders in yaw are presented. The effects of transpiration cooling, Prandtl number, pressure gradient, wall temperature, and viscosity relation were investigated. It is shown that as the Mach number is increased for a given large yaw angle the effects of pressure gradient become larger and the quantity of coolant required to maintain a given wall temperature is also increased. The use of a linear viscosity-temperature relation gives approximately the same results as the Sutherland viscosity-temperature relation except for very high aerodynamic heating rates.

Author

Heat Transfer; Skin Friction; Laminar Boundary Layer; Data Acquisition; Aerodynamic Heating

19980231023 NASA Langley Research Center, Hampton, VA USA

Theory and Apparatus for Measurement of Emissivity for Radiative Cooling of Hypersonic Aircraft with Data for Inconel X, Stainless Steel 303, and Titanium Alloy RS-120

OSullivan, William J., Jr., NASA Langley Research Center, USA; Wade, William R., NASA Langley Research Center, USA; 1961; 28p; In English

Report No.(s): NASA-TR-R-90; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The importance of radiation as a means of cooling high supersonic and hypersonic speed aircraft is discussed to show the need for measurements of the total hemispherical emissivity of surfaces. The theory underlying the measurements of the total hemispherical emissivity of surfaces is presented, readily duplicable apparatus for performing the measurements is described, and measurements for stably oxidized Inconel, Inconel X, stainless steel 303, and titanium alloy RS-120 are given for the temperature range from 600 degrees F.

Derived from text

Hypersonic Aircraft; Emissivity; Radiant Cooling; Surface Cooling; Aerodynamic Heating; Flat Plates; Radiation Measurement; Measuring Instruments; Aircraft Structures

19980231085 NASA Langley Research Center, Hampton, VA USA

A Method for Computing Turbulent Heat Transfer in the Presence of a Streamwise Pressure Gradient for Bodies in High-Speed Flow

Cohen, Nathaniel B., NASA Langley Research Center, USA; Mar. 1959; 80p; In English

Report No.(s): NASA-MEMO-1-2-59L; L-117; No Copyright; Avail: CASI; A05, Hardcopy; A01, Microfiche

A modified Reynolds analogy between skin friction and heat transfer which depends upon local pressure gradient is derived. Exact and approximate solutions are derived from the differential equations; the exact solution is applicable for arbitrary initial (transition) conditions and the approximate solution requires fully developed turbulent flow from stagnation point or leading edge. The exact solution (restricted to stagnation initial conditions) and the approximate solutions are shown to agree with one another within 5 percent when applied to several blunt shapes. The present solutions generally predict the measured heating rates on these bodies within the accuracy of the measurements provided transition began upstream of the peak heating region. The present solutions appear to be sufficiently accurate for design purposes.

Author

High Speed; Turbulent Flow; Turbulent Heat Transfer; Pressure Gradients; Computational Fluid Dynamics; Aerodynamic Heat Transfer

14
LIFE SCIENCES

Includes life sciences (general); aerospace medicine; behavioral sciences; man/system technology and life support; and space biology.

19980227853 NASA Ames Research Center, Moffett Field, CA USA

Physiological Effects of Acceleration Observed During a Centrifuge Study of Pilot Performance

Smedal, Harald A., NASA Ames Research Center, USA; Creer, Brent Y., NASA Ames Research Center, USA; Wingrove, Rodney C., NASA Ames Research Center, USA; Journal of Aerospace Medicine; Dec. 1960; Volume 31, No. 11, pp. 901-906; In English, 5-11 May 1960, Miami Beach, FL, USA

Report No.(s): NASA-TN-D-345; A-453; No Copyright; Avail: CASI; A04, Hardcopy; A01, Microfiche

An investigation was conducted by the National Aeronautics and Space Administration, Ames Research Center, and the Naval Air Development Center, Aviation Medical Acceleration Laboratory, to study the effects of acceleration on pilot performance and to obtain some meaningful data for use in establishing tolerance to acceleration levels. The flight simulator used in the study was the Johnsville centrifuge operated as a closed loop system. The pilot was required to perform a control task in various sustained acceleration fields typical of those that might be encountered by a pilot flying an entry vehicle in which he is seated in a forward-facing position. A special restraint system was developed and designed to increase the pilot's tolerance to these accelerations. The results of this study demonstrated that a well-trained subject, such as a test pilot, can adequately carry out a control task during moderately high accelerations for prolonged periods of time. The maximum levels of acceleration tolerated were approximately 6 times that of gravity for approximately 6 minutes, and varied slightly with the acceleration direction. The tolerance runs were in each case terminated by the subject. In all but two instances, the cause was extreme fatigue. On two occasions the subject terminated the run when he "grayed out." Although there were subjective and objective findings involving the visual and cardiovascular systems, the respiratory system yielded the more critical limiting factors. It would appear that these limiting factors were less severe during the "eyeballs-out" accelerations when compared with the "eyeballs-in" accelerations. These findings are explained on the basis of the influence that the inertial forces of acceleration have on the mechanics of respiration. A condensed version of this report was presented at the Annual Meeting of the Aerospace Medical Association, Miami Beach, May 5-11, 1960, in a paper entitled "Ability of Pilots to Perform a Control Task in Various Sustained Acceleration Fields."

Author

Physiological Effects; Pilot Performance; Aerospace Medicine; NASA Programs; Flight Simulators; Feedback Control; Centrifuges

16 PHYSICS

Includes physics (general); acoustics; atomic and molecular physics; nuclear and high-energy; optics; plasma physics; solid-state physics; and thermodynamics and statistical physics.

19980228155 NASA Langley Research Center, Hampton, VA USA

Transonic Performance Characteristics of Several Jet Noise Suppressors

Schmeer, James W., NASA Langley Research Center, USA; Salters, Leland B., Jr., NASA Langley Research Center, USA; Cassetti, Marlowe D., NASA Langley Research Center, USA; Jul. 1960; 50p; In English

Report No.(s): NASA-TN-D-388; L-850; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

An investigation of the transonic performance characteristics of several noise-suppressor configurations has been conducted in the Langley 16-foot transonic tunnel. The models were tested statically and over a Mach number range from 0.70 to 1.05 at an angle of attack of 0 deg. The primary jet total-pressure ratio was varied from 1.0 (jet off) to about 4.5. The effect of secondary air flow on the performance of two of the configurations was investigated. A hydrogen peroxide turbojet-engine simulator was used to supply the hot-jet exhaust. An 8-lobe afterbody with centerbody, short shroud, and secondary air had the highest thrust-minus-drag coefficients of the six noise-suppressor configurations tested. The 12-tube and 12-lobe afterbodies had the lowest internal losses. The presence of an ejector shroud partially shields the external pressure distribution of the 8-lobe after-body from the influence of the primary jet. A ring-airfoil shroud increased the static thrust of the annular nozzle but generally decreased the thrust minus drag at transonic Mach numbers.

Author

Noise Reduction; Aerodynamic Drag; Pressure Distribution; Secondary Flow; Turbojet Engines; Afterbodies; Jet Aircraft Noise

19980228190 NASA Lewis Research Center, Cleveland, OH USA

Ground Reflection of Jet Noise

Howes, Walton L., NASA Lewis Research Center, USA; 1959; 34p; In English

Report No.(s): NASA-TR-R-35; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The effect of a reflecting plane is determined from theory and experiment. From the theoretical characteristics of far-field acoustic decay a correction-to-free-field procedure is developed for data obtained in the presence of a plane. Measurements of jet noise indicated the practical significance of reflections. Several theoretical predications were confirmed from experiment decay curves and corrected spectra.

Author

Jet Aircraft Noise; Reflection; Numerical Analysis; Experimentation; Procedures

19980228210 NASA Langley Research Center, Hampton, VA USA

Preliminary Measurements of the Noise Characteristics of Some Jet-Augmented-Flap Configurations

Maglieri, Domenic J., NASA Langley Research Center, USA; Hubbard, Harvey H., NASA Langley Research Center, USA; Jan. 1959; 26p; In English

Report No.(s): NASA-MEMO-12-4-58L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

Experimental noise studies were conducted on model configurations of some proposed jet-augmented flaps to determine their far-field noise characteristics. The tests were conducted using cold-air jets of circular and rectangular exits having equal areas, at pressure ratios corresponding to exit velocities slightly below choking. Results indicated that the addition of a flap to a nozzle may change both its noise radiation pattern and frequency spectrum. Large reductions in the noise radiated in the downward direction are realized when the flow from a long narrow rectangular nozzle is permitted to attach to and flow along a large flap surface. Deflecting or turning the jet flow by means of impingement on the under surfaces increases the noise radiated in all directions and especially in the downward direction for the jet-flap configurations tested. Turning of the flow from nozzles by means of a flap turns the noise pattern approximately an equal amount. The principle of using a jet-flap shield with flow attachment may have some application as a noise suppressor.

Author

Aeroacoustics; Aerodynamic Noise; Far Fields; Frequency Distribution; Impingement; Jet Flaps; Jet Flow; Noise Measurement

19980228218 NASA Langley Research Center, Hampton, VA USA

Noise Problems Associated with Ground Operations of Jet Aircraft

Hubbard, Harvey H., NASA Langley Research Center, USA; Mar. 1959; 18p; In English

Report No.(s): NASA-MEMO-3-5-59L; No Copyright; Avail: CASI; A03, Hardcopy; A01, Microfiche

The nature of the noise-exposure problem for humans and the aircraft-structural-damage problem is each discussed briefly. Some discussion is directed toward available methods of minimizing the effects of noise on ground crews, on the aircraft structure, and on the surrounding community. A bibliography of available papers relating to noise-reduction devices is also included.

Author

Bibliographies; Noise Reduction; Aircraft Structures; Ground Crews; Jet Aircraft Noise; Noise Pollution

Subject Term Index

A

ACCELERATION, 50
ACOUSTIC EMISSION, 35
ACOUSTIC SIMULATION, 87
ACOUSTICS, 35
ACTUATORS, 15
ADAPTIVE CONTROL, 57
ADIABATIC CONDITIONS, 5
AERIAL PHOTOGRAPHY, 15
AEROACOUSTICS, 94
AERODYNAMIC BALANCE, 9, 14, 46, 56, 84
AERODYNAMIC CHARACTERISTICS, 3, 4, 6, 7, 8, 9, 10, 11, 12, 14, 17, 18, 19, 20, 21, 23, 25, 26, 33, 34, 36, 37, 40, 46, 48, 49, 57, 65, 71, 74, 79, 80, 84, 86
AERODYNAMIC COEFFICIENTS, 2, 3, 10, 11, 13, 16, 21, 22, 27, 42, 43, 46, 48, 56, 64, 74, 75, 87, 88, 91
AERODYNAMIC CONFIGURATIONS, 7, 13, 18, 26, 28, 38, 40, 58, 59, 61, 65, 75, 81, 84, 90
AERODYNAMIC DRAG, 2, 7, 9, 10, 13, 16, 21, 28, 42, 48, 56, 73, 80, 94
AERODYNAMIC FORCES, 2, 46, 62
AERODYNAMIC HEAT TRANSFER, 23, 82, 90, 93
AERODYNAMIC HEATING, 5, 10, 12, 13, 23, 38, 55, 67, 79, 81, 85, 87, 90, 92
AERODYNAMIC INTERFERENCE, 34, 87, 88
AERODYNAMIC LOADS, 5, 6, 7, 9, 13, 34, 45
AERODYNAMIC NOISE, 78, 85, 94
AERODYNAMIC STABILITY, 6, 58, 59, 61, 66, 71, 74, 77
AERODYNAMICS, 1, 16, 32, 44, 57, 82
AEROELASTICITY, 5, 6, 15, 21, 32, 57, 61
AERONAUTICAL ENGINEERING, 1, 4, 42, 84
AEROSPACE MEDICINE, 93
AEROSPACE VEHICLES, 5
AEROTHERMODYNAMICS, 23, 32
AFTERBODIES, 2, 8, 14, 28, 56, 94
AFTERBURNING, 54
AIR COOLING, 54, 55
AIR FLOW, 4, 53, 54, 82
AIR INTAKES, 13
AIR JETS, 26
AIR NAVIGATION, 36

AIRCRAFT BRAKES, 43, 89
AIRCRAFT CONFIGURATIONS, 23, 44, 66
AIRCRAFT CONSTRUCTION MATERIALS, 83
AIRCRAFT CONTROL, 22, 56
AIRCRAFT DESIGN, 1, 7, 25, 71
AIRCRAFT ENGINES, 25
AIRCRAFT LANDING, 40, 43
AIRCRAFT MANEUVERS, 58
AIRCRAFT MODELS, 7, 29, 30, 62, 65
AIRCRAFT PERFORMANCE, 39
AIRCRAFT SAFETY, 37
AIRCRAFT SPIN, 26, 65, 70
AIRCRAFT STABILITY, 62, 64, 69, 70, 71, 74
AIRCRAFT STRUCTURES, 10, 58, 61, 92, 95
AIRCRAFT TIRES, 89
AIRFOIL PROFILES, 6, 14, 16, 19, 20, 21, 22, 27, 30, 59, 65, 70, 72, 89
AIRFOILS, 3, 12, 13, 92
AIRFRAMES, 15
AIRPORTS, 36
AIRSHIPS, 20
AIRSPEED, 40, 41, 69
ALTIMETERS, 50
ALTITUDE, 79
ALTITUDE CONTROL, 72
ALTITUDE SIMULATION, 44
ALUMINUM ALLOYS, 6
ALUMINUM BOROHYDRIDES, 82
ANALOG COMPUTERS, 61, 73
ANALOG DATA, 88
ANGLE OF ATTACK, 2, 16, 18, 19, 24, 27, 29, 31, 32, 40, 61, 62, 66, 73, 75, 77, 82, 84, 91
ANGLES (GEOMETRY), 53
ANGULAR VELOCITY, 46, 69
ANNULAR NOZZLES, 51
ANTIMISSILE MISSILES, 46
APPROACH CONTROL, 43
APPROXIMATION, 47, 48
ARROW WINGS, 89
ASPECT RATIO, 12, 15, 25, 33, 40, 42, 55, 62
ATMOSPHERIC ENTRY, 5, 11, 38, 47, 81, 82
ATTITUDE (INCLINATION), 74
AUTOMATIC CONTROL, 58, 75
AUTOMATIC PILOTS, 72
AUXILIARY PROPULSION, 31

AXIAL FLOW, 52
AXISYMMETRIC BODIES, 84
AXISYMMETRIC FLOW, 76

B

B-52 AIRCRAFT, 46
B-70 AIRCRAFT, 2
BAFFLES, 44
BENDING, 62
BENDING MOMENTS, 18, 41, 45, 48
BIBLIOGRAPHIES, 1, 95
BLADE TIPS, 52, 85
BLOWING, 10, 22, 24
BLUNT BODIES, 13, 47, 84
BLUNT LEADING EDGES, 31
BOATTAILS, 2, 28
BODIES OF REVOLUTION, 18
BODY-WING AND TAIL CONFIGURATIONS, 20
BODY-WING CONFIGURATIONS, 4, 6, 9, 14, 15, 23, 24, 25, 30, 32, 58, 62, 64
BOEING 737 AIRCRAFT, 29
BOMBER AIRCRAFT, 17, 70
BOOSTER ROCKET ENGINES, 70, 81
BOOSTER ROCKETS, 9
BOOSTGLIDE VEHICLES, 38, 58
BORON ALLOYS, 83
BOTTLES, 35
BOUNDARIES, 32
BOUNDARY LAYER CONTROL, 10, 22, 24, 44
BOUNDARY LAYER FLOW, 90
BOUNDARY LAYER SEPARATION, 10, 22, 56, 70, 84
BOUNDARY LAYER TRANSITION, 2, 5, 32, 84, 85, 90, 91
BOUNDARY LAYERS, 3, 12
BOW WAVES, 28
BRAKES (FOR ARRESTING MOTION), 11
BRAKING, 89
BURNING TIME, 46

C

CAMBERED WINGS, 33, 42, 62
CANARD CONFIGURATIONS, 7, 9, 40, 42, 65, 68, 70
CAST ALLOYS, 83

CATAPULTS, 70
 CAVITATION FLOW, 91
 CENTER OF GRAVITY, 59, 64
 CENTRIFUGES, 93
 CERMETS, 55
 CESSNA AIRCRAFT, 27
 CHARTS, 10
 CHORDS (GEOMETRY), 59
 CIRCLES (GEOMETRY), 28
 CIRCULAR CONES, 32
 CIRCULAR CYLINDERS, 14, 16, 85
 CIRCULAR ORBITS, 77, 80
 CLIMBING FLIGHT, 41
 COEFFICIENTS, 14, 49
 COLLISION PARAMETERS, 87
 COMBUSTION CHAMBERS, 35, 53, 54
 COMMERCIAL AIRCRAFT, 35
 COMPOSITE MATERIALS, 79
 COMPRESSIBLE FLOW, 59
 COMPRESSOR EFFICIENCY, 53
 COMPRESSOR ROTORS, 52, 53
 COMPUTATION, 21, 48
 COMPUTATIONAL FLUID DYNAMICS, 52, 93
 CONCAVITY, 90
 CONGRESSIONAL REPORTS, 37
 CONICAL BODIES, 20, 26, 84
 CONICAL CAMBER, 25, 62
 CONICAL FLOW, 26
 CONICAL NOZZLES, 86
 CONTROL, 68
 CONTROL STABILITY, 56, 69
 CONTROL STICKS, 68
 CONTROL SURFACES, 67, 71, 73
 CONTROL SYSTEMS DESIGN, 15, 39, 69, 72, 78
 CONTROL THEORY, 15, 74
 CONTROLLABILITY, 26, 46, 65, 73
 CONTROLLERS, 58, 73
 CONVECTIVE HEAT TRANSFER, 3, 5, 81
 CONVERGENT NOZZLES, 4, 76
 COOLANTS, 55
 COOLING, 86
 COOLING SYSTEMS, 55
 COST REDUCTION, 81
 COUPLED MODES, 32, 43
 COWLINGS, 76
 CRACK INITIATION, 53
 CRASHES, 35
 CRITICAL PRESSURE, 28
 CROSS FLOW, 11, 84, 85, 89
 CYLINDRICAL BODIES, 4, 20, 43, 56

D

DAMPING, 23, 47, 72
 DATA ACQUISITION, 8, 19, 29, 30, 31, 43, 51, 86, 92
 DATA FLOW ANALYSIS, 72
 DATA PROCESSING EQUIPMENT, 2
 DATA SYSTEMS, 29
 DECELERATION, 11, 40
 DECISIONS, 36
 DEFLECTION, 34, 65
 DEGREES OF FREEDOM, 57
 DELTA WINGS, 13, 23, 25, 27, 29, 42, 46, 49, 62, 80, 89
 DENSITY (MASS/VOLUME), 19
 DESCENT, 41, 69
 DESIGN ANALYSIS, 25, 52, 72, 78, 81
 DETECTION, 20
 DETONATION WAVES, 31
 DIALS, 50
 DIFFERENTIAL EQUATIONS, 18, 49
 DIFFRACTION, 69
 DIGITAL SYSTEMS, 72
 DIHEDRAL ANGLE, 23, 91
 DIMENSIONAL STABILITY, 23
 DIRECTIONAL CONTROL, 61, 71, 77
 DIRECTIONAL STABILITY, 9, 23, 27, 31, 38, 46, 48, 57, 63, 69, 72
 DIVERGENCE, 15
 DIVERTERS, 3, 13
 DOWNWASH, 2
 DRAG, 11, 29
 DRAG MEASUREMENT, 21
 DRAG REDUCTION, 7, 28, 42
 DRONE VEHICLES, 68, 70
 DUCTED FANS, 60
 DYNAMIC CHARACTERISTICS, 19, 34, 61, 68, 77
 DYNAMIC CONTROL, 2, 39, 66
 DYNAMIC MODELS, 35, 43, 70
 DYNAMIC PRESSURE, 78
 DYNAMIC STABILITY, 23, 31, 39, 47, 62, 63, 66, 70, 74
 DYNAMIC TESTS, 60

E

EARTH ATMOSPHERE, 77, 82
 EARTH ORBITAL RENDEZVOUS, 74
 EARTH ORBITS, 74, 81
 EDUCATION, 37, 38
 EFFECTIVENESS, 33
 EFFICIENCY, 52
 EJECTION, 44
 EJECTORS, 28
 ELASTIC PROPERTIES, 6

EMISSION, 92
 END PLATES, 12, 44
 ENERGY DISSIPATION, 80
 ENGINE AIRFRAME INTEGRATION, 44
 ENGINE DESIGN, 44, 53
 ENGINE FAILURE, 53
 ENGINE INLETS, 13, 28, 55
 ENGINE PARTS, 53
 ENGINE TESTS, 53
 ENVIRONMENTAL TESTS, 79
 EQUATIONS OF MOTION, 64, 74
 EQUATORIAL ORBITS, 80
 ERRORS, 11, 50
 ESTIMATES, 63
 ESTIMATING, 28, 38
 EVALUATION, 19, 48, 55, 63, 70
 EXAMINATION, 61
 EXHAUST GASES, 34
 EXPERIMENTATION, 9, 16, 20, 25, 29, 30, 33, 34, 43, 50, 52, 55, 67, 75, 86, 88, 94
 EXTERNALLY BLOWN FLAPS, 10, 22, 24

F

F-100 AIRCRAFT, 45
 F-106 AIRCRAFT, 44
 F-15 AIRCRAFT, 42
 FABRICATION, 76
 FAILURE, 55
 FAIRINGS, 2
 FAR FIELDS, 94
 FEEDBACK CONTROL, 93
 FIGHTER AIRCRAFT, 38, 46, 58, 59
 FINENESS RATIO, 12, 14, 28, 63
 FINS, 63, 91
 FIRE EXTINGUISHERS, 35
 FIRE PREVENTION, 35
 FIRES, 35
 FLAME HOLDERS, 54
 FLAME STABILITY, 54
 FLAPPING, 6, 10, 18, 19, 26, 27, 71
 FLAPPING HINGES, 18
 FLAPS (CONTROL SURFACES), 44, 45
 FLAT PLATES, 3, 24, 84, 92
 FLEXIBLE WINGS, 6, 13, 21
 FLICKER, 73
 FLIGHT CHARACTERISTICS, 38, 43, 60, 62, 72, 81
 FLIGHT CONDITIONS, 41, 71, 72, 81, 91
 FLIGHT CONTROL, 57, 58, 69
 FLIGHT INSTRUMENTS, 78
 FLIGHT LOAD RECORDERS, 29

FLIGHT OPERATIONS, 58
 FLIGHT PATHS, 38, 40, 46, 49, 77
 FLIGHT SAFETY, 37
 FLIGHT SIMULATORS, 76, 93
 FLIGHT TESTS, 8, 31, 39, 41, 44, 50, 52, 57, 58, 69, 71, 75, 85, 88, 90
 FLIGHT TIME, 38
 FLOW CHARACTERISTICS, 12, 17, 89
 FLOW DISTORTION, 33
 FLOW DISTRIBUTION, 1, 3, 24, 53, 88, 90
 FLOW THEORY, 25
 FLOW VELOCITY, 85
 FLOW VISUALIZATION, 48
 FLUTTER, 15, 51, 59, 75
 FLUTTER ANALYSIS, 32, 59, 61, 67
 FORCE DISTRIBUTION, 2
 FOREBODIES, 58, 65, 73
 FORMAT, 51
 FREE FLIGHT, 58, 61, 66, 73, 80, 89, 90, 91
 FREE FLOW, 6, 11, 12, 13, 40, 84, 90
 FREE JETS, 6, 85, 87
 FREE VIBRATION, 92
 FREQUENCIES, 38, 43
 FREQUENCY DISTRIBUTION, 94
 FRICTION, 61, 89
 FRICTION DRAG, 3
 FRICTION MEASUREMENT, 19
 FULL SCALE TESTS, 86
 FUNCTIONAL ANALYSIS, 59
 FUSELAGES, 1, 2, 30, 35, 65, 73

G

GAS FLOW, 75
 GAS GENERATORS, 4, 76
 GAS TURBINE ENGINES, 83
 GEOSTROPHIC WIND, 77
 GLIDE PATHS, 73
 GLIDERS, 57
 GLIDING, 91
 GRAPHS (CHARTS), 36
 GRAVITATIONAL FIELDS, 80
 GROUND BASED CONTROL, 39
 GROUND CREWS, 95
 GROUND EFFECT (AERODYNAMICS), 6, 12, 14, 33
 GUIDANCE (MOTION), 67
 GYROSCOPES, 78
 GYROSCOPIC STABILITY, 64

H

HARMONIC ANALYSIS, 18

HEAT RESISTANT ALLOYS, 79
 HEAT SHIELDING, 79
 HEAT SINKS, 82
 HEAT TRANSFER, 19, 20, 85, 86, 89, 91, 92
 HEAT TRANSFER COEFFICIENTS, 32, 87
 HEATING, 3, 84
 HELICOPTER PERFORMANCE, 85
 HELICOPTERS, 22, 85
 HELIUM, 75
 HIGH ACCELERATION, 45
 HIGH ALTITUDE, 73
 HIGH ASPECT RATIO, 10, 75
 HIGH SPEED, 93
 HIGH STRENGTH ALLOYS, 83
 HIGH TEMPERATURE GASES, 28
 HIGH TEMPERATURE TESTS, 10
 HOMING, 67
 HORIZONTAL FLIGHT, 31, 39, 60
 HORIZONTAL TAIL SURFACES, 55, 68
 HORSESHOE VORTICES, 68
 HOT-WIRE ANEMOMETERS, 85
 HOVERING, 22, 31, 39, 88
 HUBS, 21
 HYDRODYNAMICS, 87, 91
 HYDROFOILS, 87, 91
 HYDROGEN, 76
 HYDROGEN PEROXIDE, 4
 HYPERSONIC AIRCRAFT, 92
 HYPERSONIC FLOW, 31, 92
 HYPERSONIC GLIDERS, 13, 64
 HYPERSONIC SPEED, 13, 20, 31, 75, 90, 92
 HYPERSONIC VEHICLES, 58
 HYPERSONICS, 20, 91

I

IGNITION, 35
 IMPACT DAMAGE, 83
 IMPACT LOADS, 88
 IMPACT RESISTANCE, 83
 IMPACT TESTS, 83, 88
 IMPINGEMENT, 94
 IN-FLIGHT MONITORING, 72
 INCOMPRESSIBLE FLOW, 53
 INDEXES (DOCUMENTATION), 1
 INDUCED DRAG, 12, 43
 INERTIA, 47
 INLET FLOW, 33, 71
 INLET NOZZLES, 33
 INLET TEMPERATURE, 54, 55
 INSTALLING, 83

INSULATED STRUCTURES, 79
 INSULATION, 79
 INTEGRAL EQUATIONS, 33
 INTERACTIONAL AERODYNAMICS, 27
 INTERCONTINENTAL BALLISTIC MISSILES, 46, 85
 INTERFERENCE DRAG, 34
 ITERATION, 6

J

JET AIRCRAFT, 18, 36, 37, 40, 70
 JET AIRCRAFT NOISE, 94, 95
 JET BOUNDARIES, 91
 JET ENGINES, 43, 86
 JET EXHAUST, 4, 28, 76
 JET FLAPS, 24, 26, 94
 JET FLOW, 10, 85, 94
 JET LIFT, 26
 JP-4 JET FUEL, 79

K

KERNEL FUNCTIONS, 2, 59

L

LAMINAR BOUNDARY LAYER, 40, 85, 89, 91, 92
 LAMINAR FLOW, 12, 23, 85, 91
 LANDING, 50, 75
 LANDING GEAR, 80
 LANDING SPEED, 37
 LATERAL CONTROL, 38, 56, 60, 61, 70, 71, 72
 LATERAL STABILITY, 6, 9, 23, 25, 44, 58, 60, 61, 64, 68, 71
 LAUNCHING, 46
 LEADING EDGE FLAPS, 22, 25
 LEADING EDGE SWEEP, 24, 28, 59, 85
 LEADING EDGES, 17, 21, 23, 24, 30, 31, 40, 53, 55
 LENGTH, 65
 LENTICULAR BODIES, 79
 LIFT, 10, 13, 14, 22, 30, 33, 48
 LIFT AUGMENTATION, 24
 LIFT DEVICES, 26
 LIFT DRAG RATIO, 3, 8, 9, 10, 11, 14, 19, 24, 25, 30, 40, 42, 48, 50, 62, 73, 77, 91
 LIFTING BODIES, 5, 8
 LIFTING ROTORS, 27, 91
 LIGHT AIRCRAFT, 69
 LINEARITY, 12

LININGS, 53
 LIQUID OXYGEN, 79
 LOAD DISTRIBUTION (FORCES), 15, 89
 LOADS (FORCES), 29, 61, 68
 LONGITUDINAL CONTROL, 40, 56, 57, 58, 59, 61, 71, 80
 LONGITUDINAL STABILITY, 9, 15, 16, 25, 27, 30, 48, 56, 58, 60, 64, 65, 66, 68, 70, 71, 72, 73, 80
 LOW ASPECT RATIO, 17, 63
 LOW ASPECT RATIO WINGS, 10, 15, 24, 33, 44, 73
 LOW PRESSURE, 9
 LOW SPEED, 16, 20, 37, 75
 LUBRICATING OILS, 83

M

MACH NUMBER, 3, 4, 10, 16, 29, 42, 67, 73, 75, 84
 MAGNITUDE, 4
 MANNED REENTRY, 77, 78, 80
 MANNED SPACECRAFT, 81
 MATRICES (MATHEMATICS), 13
 MATRIX METHODS, 13
 MEASURING INSTRUMENTS, 43, 72, 92
 MECHANICAL PROPERTIES, 83
 METAL COMBUSTION, 82
 MILITARY HELICOPTERS, 41
 MILITARY OPERATIONS, 41
 MILITARY VEHICLES, 34
 MINIMUM DRAG, 3
 MISALIGNMENT, 47
 MISSILE BODIES, 11
 MISSILE CONFIGURATIONS, 44
 MISSILE CONTROL, 67, 69
 MISSILES, 63
 MODELS, 15, 18, 19, 20, 75
 MOMENT DISTRIBUTION, 56
 MOMENTS, 15
 MOTION SIMULATION, 76
 MULTIENGINE VEHICLES, 36

N

NACELLES, 2, 28
 NASA PROGRAMS, 93
 NATIONAL AIRSPACE SYSTEM, 36
 NAVAHO MISSILE, 9
 NAVIGATION, 67
 NICKEL ALLOYS, 83
 NITROGEN, 28
 NOISE (SOUND), 65

NOISE INTENSITY, 30
 NOISE MEASUREMENT, 78, 94
 NOISE POLLUTION, 95
 NOISE REDUCTION, 94, 95
 NONDESTRUCTIVE TESTS, 35
 NONLINEARITY, 68
 NORMAL DENSITY FUNCTIONS, 38
 NOSE TIPS, 9
 NOSES (FOREBODIES), 90
 NOZZLE EFFICIENCY, 52
 NOZZLE FLOW, 51
 NUMERICAL ANALYSIS, 15, 30, 43, 49, 64, 67, 86, 94

O

OGIVES, 56
 OPERATIONAL PROBLEMS, 36
 OPTIMAL CONTROL, 26, 67
 OPTIMIZATION, 49
 ORBITAL MECHANICS, 74
 OSCILLATIONS, 65
 OXYGEN, 28

P

PARAGLIDERS, 19, 28
 PARASITES, 21
 PAYLOADS, 11, 81
 PERFORMANCE TESTS, 31, 75
 PERIGEEES, 74
 PERSONNEL DEVELOPMENT, 37
 PERTURBATION, 3
 PHOTOGRAPHS, 90
 PHYSIOLOGICAL EFFECTS, 93
 PIEZOELECTRIC CERAMICS, 15
 PILOT INDUCED OSCILLATION, 56
 PILOT PERFORMANCE, 93
 PILOT TRAINING, 76
 PILOTLESS AIRCRAFT, 63
 PITCH (INCLINATION), 44, 48, 49, 52
 PITCHING MOMENTS, 4, 24, 25, 29, 30, 39, 62, 67, 71, 75, 91
 PLANFORMS, 28
 POSITION ERRORS, 9
 POSITIONING, 50
 POTENTIAL FLOW, 1, 3
 POWERED MODELS, 76
 PREDICTIONS, 1
 PRESSURE DISTRIBUTION, 1, 13, 15, 16, 21, 28, 31, 32, 33, 34, 48, 82, 88, 90, 91, 94
 PRESSURE DRAG, 3, 10, 28, 76
 PRESSURE GRADIENTS, 5, 93
 PRESSURE MEASUREMENT, 7, 17, 85

PRESSURE RATIO, 52
 PRESSURE REDUCTION, 53
 PRESSURE SENSORS, 6, 11
 PROCEDURES, 21, 48, 49, 83, 94
 PROPAGATION VELOCITY, 3
 PROPELLER SLIPSTREAMS, 6, 27
 PROPELLERS, 90
 PROPULSION, 84
 PYLONS, 3

R

RADAR HOMING MISSILES, 69
 RADIANT COOLING, 92
 RADIATION MEASUREMENT, 92
 RADIATIVE HEAT TRANSFER, 5
 RADOMES, 69
 RAMJET ENGINES, 68, 70
 RECTANGULAR WINGS, 10, 23, 25, 26, 33, 61, 89
 REENTRY EFFECTS, 81
 REENTRY VEHICLES, 77, 78, 79, 80, 81, 86
 REFLECTION, 94
 RESEARCH, 29, 33, 49, 67, 83, 87
 RESEARCH AIRCRAFT, 38
 RESEARCH VEHICLES, 4
 REYNOLDS NUMBER, 10, 16, 30, 32, 75
 RIGID STRUCTURES, 48
 RIGID WINGS, 28
 ROCKET ENGINE NOISE, 78
 ROCKET ENGINES, 79
 ROCKET EXHAUST, 79
 ROCKET VEHICLES, 56, 78
 ROLL, 45, 58, 68
 ROLLING MOMENTS, 67, 73
 ROTARY STABILITY, 62
 ROTARY WINGS, 7, 13, 21, 22, 43, 44, 45, 85
 ROTATING BODIES, 33
 ROTATING STALLS, 48
 ROTATION, 26
 ROTOR AERODYNAMICS, 27, 48, 49
 ROTOR LIFT, 49
 ROTORS, 7, 18, 33, 48, 49, 55, 85
 RUDDERS, 77
 RUNWAY CONDITIONS, 45, 47
 RUNWAYS, 40, 43, 45, 47, 50, 89

S

SAFETY MANAGEMENT, 36
 SAMPLED DATA SYSTEMS, 72
 SATELLITE ATTITUDE CONTROL, 78

SATELLITE INSTRUMENTS, 78
 SCALE MODELS, 2, 9, 13, 15, 26, 27,
 33, 34, 39, 44, 46, 58, 59, 60, 65, 66,
 68, 70, 71, 72, 75, 85
 SEAPLANES, 25
 SECONDARY FLOW, 94
 SEMISPAN MODELS, 27
 SEPARATED FLOW, 10, 56
 SHADOWGRAPH PHOTOGRAPHY, 28
 SHAPERS, 43
 SHOCK WAVES, 28, 31
 SHORT TAKEOFF AIRCRAFT, 44
 SIDESLIP, 27, 58, 68
 SIGNAL GENERATORS, 57
 SIMULATION, 4, 25
 SIMULATORS, 4, 68, 76
 SINE WAVES, 57
 SKIN FRICTION, 5, 92
 SLENDER BODIES, 4, 18
 SLENDER WINGS, 15
 SLIDING FRICTION, 89
 SLOPES, 24
 SMOKE, 88
 SPACE MISSIONS, 77
 SPACE SHUTTLE BOOSTERS, 81
 SPACECRAFT CONFIGURATIONS, 80
 SPACECRAFT LANDING, 79
 SPECIMENS, 49
 SPECTRUM ANALYSIS, 38
 SPHERICAL COORDINATES, 80
 SPIN DYNAMICS, 26, 65, 70
 SPIN REDUCTION, 64
 SPIN STABILIZATION, 90
 SPIN TESTS, 27, 65
 STABILITY, 8, 28, 56, 60, 65, 71
 STABILITY AUGMENTATION, 69
 STABILITY DERIVATIVES, 58, 61, 63,
 64, 68, 69, 77
 STABILITY TESTS, 47
 STABILIZATION, 54, 56
 STABILIZERS (FLUID DYNAMICS),
 15
 STAGNATION POINT, 3
 STAGNATION PRESSURE, 7, 32
 STANTON NUMBER, 32
 STATIC AERODYNAMIC CHARACTERISTICS,
 5, 21, 29, 30
 STATIC PRESSURE, 6, 9, 11
 STATIC STABILITY, 2, 7, 24, 27, 56, 57,
 58, 59, 62, 63, 65, 66, 70, 72, 73, 75,
 77, 80
 STATIC TESTS, 51, 60
 STATIC THRUST, 51
 STATOR BLADES, 55
 STATORS, 55
 STEADY FLOW, 90

STEADY STATE, 13
 STIFFENING, 6
 STIFFNESS, 87
 STOPPING, 35
 STRAIN GAGES, 51
 STRAKES, 26
 STRESS MEASUREMENT, 45
 STRESSED-SKIN STRUCTURES, 38
 STRUCTURAL ANALYSIS, 6, 45, 55,
 61, 62, 92
 STRUCTURAL INFLUENCE COEFFICIENTS, 21
 STRUCTURAL VIBRATION, 92
 STRUTS, 80
 SUBSONIC FLOW, 59, 85
 SUBSONIC SPEED, 2, 7, 13, 17, 22, 27,
 30, 47, 60, 61, 62, 63, 64, 69, 80, 90
 SUPERCAVITATING FLOW, 87
 SUPERSONIC AIRCRAFT, 42, 70
 SUPERSONIC FLIGHT, 91
 SUPERSONIC FLOW, 11, 19, 21, 84, 87
 SUPERSONIC HEAT TRANSFER, 32
 SUPERSONIC JET FLOW, 55
 SUPERSONIC NOZZLES, 87
 SUPERSONIC SPEED, 2, 3, 6, 9, 10, 15,
 18, 19, 24, 25, 30, 38, 42, 45, 55, 57,
 58, 61, 63, 66, 68, 69, 82, 84, 85, 89,
 90, 91
 SUPERSONIC TRANSPORTS, 43
 SURFACE COOLING, 92
 SURFACE PROPERTIES, 89
 SURFACE ROUGHNESS, 45, 47, 91
 SWEEP EFFECT, 84
 SWEEPBACK, 14
 SWEPT WINGS, 3, 10, 16, 23, 24, 33,
 57, 59, 84, 85, 91
 SWEPTBACK WINGS, 7, 16, 18, 22, 23,
 24, 32, 61, 62, 64, 75, 89
 SYMMETRICAL BODIES, 47

T

TAIL ASSEMBLIES, 66, 72
 TAIL SURFACES, 59
 TAKEOFF, 17, 43, 50
 TANGENTIAL BLOWING, 14
 TAPERING, 59
 TAXIING, 45
 TECHNOLOGY ASSESSMENT, 52
 TECHNOLOGY UTILIZATION, 83
 TEETERING, 45
 TEMPERATURE EFFECTS, 55, 83, 89
 TEMPERATURE MEASUREMENT, 20
 TEMPERATURE RATIO, 89
 THERMAL ENVIRONMENTS, 38
 THERMAL FATIGUE, 10, 53

THERMAL RESISTANCE, 79
 THERMOCOUPLES, 54
 THICKNESS, 75, 89
 THICKNESS RATIO, 42
 THIN AIRFOILS, 21, 42
 THIN WINGS, 17, 42, 84, 87
 THREE DIMENSIONAL FLOW, 53, 59
 THROTTLING, 75
 THRUST REVERSAL, 43
 THRUST-WEIGHT RATIO, 81
 TILT WING AIRCRAFT, 39, 66
 TILTED PROPELLERS, 44
 TIP SPEED, 85
 TITAN, 85
 TORQUE, 78
 TORSION, 87
 TORSIONAL STRESS, 41
 TOUCHDOWN, 40
 TRAILING EDGE FLAPS, 10, 22, 24, 44
 TRAILING EDGES, 10, 54
 TRAJECTORIES, 82
 TRAJECTORY CONTROL, 74, 82
 TRANSFER ORBITS, 81
 TRANSITION FLIGHT, 39
 TRANSITION POINTS, 20
 TRANSONIC COMPRESSORS, 52
 TRANSONIC FLIGHT, 8
 TRANSONIC FLOW, 52
 TRANSONIC SPEED, 6, 7, 13, 15, 23,
 30, 36, 46, 59, 68, 88
 TRANSPIRATION, 86
 TRANSPORT AIRCRAFT, 36, 37
 TRANSVERSE LOADS, 42
 TRAPEZOIDAL WINGS, 5, 65, 66
 TURBINE BLADES, 53, 55, 86
 TURBOCOMPRESSORS, 52, 53
 TURBOFAN ENGINES, 42, 51
 TURBOJET ENGINES, 2, 4, 17, 53, 54,
 83, 94
 TURBOMACHINE BLADES, 7
 TURBOPROP AIRCRAFT, 40
 TURBULENCE, 57, 85
 TURBULENT BOUNDARY LAYER, 5,
 90
 TURBULENT FLOW, 12, 85, 93
 TURBULENT HEAT TRANSFER, 93
 TWISTED WINGS, 7, 16, 42
 TWO DIMENSIONAL FLOW, 13, 84
 TWO DIMENSIONAL MODELS, 16, 28
 TWO STAGE TURBINES, 52

U

UH-2 HELICOPTER, 35
 UNSTEADY FLOW, 12, 21, 90

UNSWEPT WINGS, 10, 44, 66, 68, 73

Z

ZERO ANGLE OF ATTACK, 26

V

VARIABILITY, 19

VELOCITY, 27, 49, 50

VELOCITY DISTRIBUTION, 48

VELOCITY MEASUREMENT, 85

VENTILATION, 87

VERTICAL FLIGHT, 44

VERTICAL LANDING, 17, 41, 60, 66,
71

VERTICAL TAKEOFF AIRCRAFT, 6,
27, 31, 39, 44, 74

VIBRATION, 42

VIBRATION DAMPING, 49

VISCOUS FLOW, 84

VORTEX GENERATORS, 25

VORTICES, 30, 87, 90

VZ-2 AIRCRAFT, 39

W

WAKES, 88

WATER LANDING, 79, 86

WAVE DRAG, 18

WEATHER, 37

WEDGES, 3, 21

WIND TUNNEL MODELS, 4, 8, 57, 76

WIND TUNNEL STABILITY TESTS, 7

WIND TUNNEL TESTS, 3, 4, 6, 7, 10,
11, 12, 13, 15, 17, 18, 19, 20, 22, 23,
24, 25, 26, 27, 29, 30, 32, 33, 34, 45,
46, 47, 48, 52, 55, 57, 58, 61, 63, 64,
65, 66, 68, 71, 75, 82, 90

WING LOADING, 71, 92

WING OSCILLATIONS, 62

WING PANELS, 71

WING PLANFORMS, 3, 10

WING SPAN, 33

WING TIPS, 9, 62, 80

WINGS, 2, 4, 6, 12, 17, 20, 57, 62, 75, 82,
92

WIRE CLOTH, 86

X

X WING ROTORS, 17

X-15 AIRCRAFT, 56

Y

YAW, 58, 72

YAWING MOMENTS, 64, 67

Personal Author Index

A

Abdalla, Kaleel L., 31
Adams, Gaynor J., 42
Aiken, William S., Jr., 13
Alford, William L., 58
Altermann, John A., III, 13
Anderson, Bernhard H., 3
Anderson, Seth B., 42
Aoyagi, Kiyoshi, 23
Assadourian, Arthur, 68
Ayers, Theodore G., 70

B

Bachkin, Daniel, 52
Bandettini, Angelo, 89
Batterson, Sidney A., 89
Beattie, A. G., 35
Beckwith, Ivan E., 92
Bell, B. Ann, 28
Bellman, Donald R., 8
Bennett, Floyd V., 48, 57
Betts, John, Jr., 78
Blackaby, James R., 13
Blanchard, Ulysse J., 47, 79
Bland, William M., Jr., 12
Boisseau, Peter C., 73
Boissevain, Alfred G., 74
Booth, Katherine W., 67
Bowman, James S., 70
Bowman, James S., Jr., 20, 26, 27
Boxer, Emanuel, 32
Boyd, John W., 42, 62
Brinich, Paul F., 20
Brinkworth, Helen S., 47
Brooks, George W., 17, 43
Brown, Clarence A., Jr., 46
Brown, Lawrence W., 64
Brown, Stuart C., 21
Buell, Donald A., 61, 63, 65
Buglia, James J., 47
Busch, Arthur M., 35
Butchart, Stanley P., 35, 37
Butler, James K., 62
Butze, Helmut F., 54

C

Calvert, Howard F., 54
Campbell, John A., 35
Capone, Francis J., 6, 11
Carpenter, Gene T., 50
Carter, Arthur W., 12
Cassetti, Marlowe D., 70, 94
Cattarius, Jens, 14
Champine, Robert A., 75
Chapman, Dean R., 5
Chapman, Gary T., 26, 88

Chapman, Gray T., 84
Charczenko, Nickolai, 11
Chernyi, G. G., 31
Chiarito, Patrick T., 55
Christopher, Kenneth W., 91
Church, James D., 8
Churchill, Gary B., 21
Cicala, Placido, 48
Cochran, Reeves P., 53
Cohen, Nathaniel B., 93
Collie, Katherine A., 12
Coltrane, Lucille C., 56
Conners, Timothy R., 41
Connor, Andrew B., 41
Cooney, T. V., 72
Cooper, George E., 42
Cooper, Morton, 23
Corson, Blake W., Jr., 51
Crane, Harold L., 59, 60
Creager, Marcus O., 13, 24
Creer, Brent Y., 76, 93
Cunningham, Herbert J., 1, 15
Czarnecki, K. R., 2

D

DAiutolo, Charles T., 59
Daugherty, James C., 2
Davidson, John R., 38
Day, Richard E., 38
Dengler, Robert P., 53
DiCamillo, Joseph R., 65
Dicus, John H., 51
Dorsch, Robert G., 82
Douvillier, Joseph G., Jr., 76
Driver, Cornelius, 64
Dryer, Murray, 69
Dugan, Duane W., 66

E

Edwards, Frederick G., 46
Edwards, George G., 22
Eggleston, John M., 81
Ellington, Rex R., 67
Elliott, Miriam H., 15, 89
Emerson, Horace F., 23
Evans, David G., 51

F

Fahey, Russell E., 34
Faye, Alan E., Jr., 42
Ferguson, Michael D., 52
Ferrari, Carlo, 92
Fichter, Ann B., 25
Fink, Marvin P., 27
Fink, Marvin P., 33

Fischel, Jack, 37
Fischel, Jack, 35
Fisher, Lewis R., 47
Fisher, Lloyd J., Jr., 80
Fletcher, Edward A., 82
Fobes, J. L., 37
Fortini, Anthony, 79
Foster, Gerald V., 58
Fournier, Paul G., 7, 28
Freche, John C., 83
Funk, Jack, 72

G

Gainer, Patrick A., 13
Gainer, Thomas G., 26
Geller, Edward W., 1
Gertsma, Laurence W., 28
Gibson, Frederick W., 15
Gillis, Clarence L., 59
Goldberg, Theodore J., 32
Goodman, George P., 78
Goodson, Kenneth W., 29
Goodwin, Frederick K., 69, 79
Goodwin, Glen, 81
Gracey, William, 9, 50
Graham, John B., Jr., 85
Grant, Frederick C., 3, 30
Grantham, William D., 20, 64
Gray, Robin B., 87
Green, Kendal H., 2
Griffin, Roy N., Jr., 10
Grigsby, Carl E., 87
Grobman, Jack S., 53
Grumondz, T. A., 30
Grunwald, Kalman J., 1, 5
Guy, Lawrence D., 67

H

Hall, Albert W., 43, 47
Hamer, Harold A., 38
Hansen, Q. Marion, 78
Hanson, Perry W., 32
Harper, Eleanor V., 71
Harrin, Eziaslav N., 89
Harrington, Robert D., 21
Harris, Jack E., 56
Harris, Robert S., Jr., 38
Hassell, James L., Jr., 57, 72
Hasson, Dennis F., 24
Hathaway, Melvin E., 86
Hathaway, Ross, 52
Healy, Frederick M., 26, 27
Healy, T. J., Jr., 81
Heath, Atwood R., Jr., 7, 18
Hedgepeth, John M., 49
Henderson, Arthur, Jr., 83
Henderson, William P., 14

Hendrix, Charles D., 79
 Henning, Allen B., 63
 Hewes, Donald E., 61, 72
 Heyson, Harry H., 27, 85, 91
 Hickey, David H., 23
 Higdon, Donald T., 62
 Hilton, David A., 78
 Hinson, William F., 44
 Holzhauser, Curt A., 10, 44
 Hopkins, Edward J., 84, 89
 Howard, Charles D., 85
 Howes, Walton L., 94
 Hubbard, Harvey H., 78, 94, 95
 Huff, Vearl N., 79
 Hurrell, Herbert G., 51

I

Inman, Daniel J., 14

J

James, Carlton S., 90
 Jaquet, Byron M., 3
 Jewel, Joseph W., Jr., 27, 50, 72
 Jillie, Don W., 84
 Johnson, Joseph L., Jr., 18
 Johnson, Norman S., 63
 Johnson, Virgil E., Jr., 87, 91
 Johnston, James R., 53, 55, 82
 Johnston, Patrick J., 20
 Jones, Charles K., 69
 Jorgensen, Leland H., 8

K

Kaattari, George E., 63, 69
 Katzen, Elliott D., 11
 Keating, Stephen J., Jr., 89
 Kelly, Thomas C., 5, 34
 Keynton, Robert J., 56
 Kirk, Donn B., 26
 Klinar, Walter J., 64
 Koenig, David G., 27
 Kolnick, Joseph J., 50
 Kordes, Eldon E., 67, 92
 Krasilshchikova, E. A., 12
 Kriebel, Anthony R., 3
 Kroll, Wilhelmina D., 10
 Kruszewski, Edwin T., 42
 Kurkov, Anatole P., 51
 Kussoy, Marvin I., 52

L

Ladson, Charles L., 13, 77
 Landrum, Emma Jean, 2
 Lastinger, James L., 33
 Laufer, Th., 49
 Laurence, James C., 85
 Lawrence, George F., 47
 Lazzeroni, Frank A., 25

Lee, Henry A., 65
 Lee, John B., 44
 Lee, Louise P., 87
 Lessing, Henry C., 62
 Levine, Jack, 90
 Levy, Lionel L., Jr., 11, 18
 Lichtenstein, Jacob H., 47, 77
 Lina, Lindsay J., 75
 Lockwood, Vernard E., 14, 16
 Lomax, Harvard, 84
 Lopez, Armando E., 61
 Love, Eugene S., 87
 Lubomski, Joseph F., 51
 Ludi, LeRoy H., 41
 Luidens, Roger W., 40
 Lundstrom, Reginald R., 73
 Lyman, E. Gene, 13

M

Maglieri, Domenic J., 17, 94
 Maki, Ralph L., 21
 Margolis, Kenneth, 15, 89
 Marker, Ralph D., 34
 Martin, Norman J., 33
 Matranga, Gene J., 45
 Matting, Fred W., 5
 Mayer, John P., 38
 Mayes, William H., 78
 Mayo, Alton P., 45
 McCarty, John Locke, 17
 McGehee, John R., 86, 87
 McKann, Robert E., 47
 McKee, John W., 7
 McKinney, Linwood W., 16
 McLemore, H. Clyde, 19
 McShera, John T., 11
 Mehalic, Charles M., 51
 Menees, Gene P., 24, 62, 66
 Mercer, Charles E., 2, 28, 51
 Merlet, Charles F., 31
 Mersman, William A., 79
 Metzler, Allen J., 54
 Meyer, Andre J., Jr., 54
 Miller, Robert W., 75
 Milwitzky, Benjamin, 45
 Mitchell, Jesse L., 59
 Mixson, John S., 88
 Monakhov, N. M., 33
 Monroe, Daniel E., 52, 55
 Montgomery, John C., 51
 Morgan, Homer G., 61, 75
 Morgan, James R., 25
 Morgan, William C., 54
 Morris, Garland J., 47, 75
 Moseley, William C., Jr., 17
 Moul, Martin T., 64
 Mugler, John P., Jr., 6, 7, 16, 32

N

Naeseth, Rodger L., 7
 Neiderman, Eric C., 37
 Nelson, Richard D., 81

Nelson, Robert L., 4
 Newsom, William A., Jr., 71
 Nielsen, Jack N., 79
 North, Warren J., 69
 Norton, Harry T., Jr., 28
 Nyholm, Jack R., 5

O

Orme, John S., 52
 Ostoslavskii, I. V., 30
 OSullivan, William J., Jr., 92

P

Panov, V. V., 83
 Parlett, Lysle P., 60
 Paulson, John W., 57, 59, 73, 75
 Pearson, Albin O., 29, 47
 Peterson, John B., Jr., 19
 Peterson, Victor L., 9, 66
 Pinkel, I. Irving, 82
 Polhamus, Edward C., 1
 Powell, Robert D., Jr., 22
 Pratt, Kermit G., 57
 Presnell, John G., Jr., 24

Q

Queijo, M. J., 57

R

Ragsdale, Robert G., 19
 Rathert, George A., Jr., 76
 Rebont, Jean, 48, 49
 Redont, Jean, 43
 Reeder, John P., 49
 Reisert, Donald, 38
 Richards, Hadley T., 86
 Rind, Emanuel, 50
 Ritchie, Virgil S., 9
 Robinson, Glenn H., 35, 37
 Robinson, Robert C., 23, 46
 Rolls, L. Stewart, 29
 Romanov, G. L., 74
 Rosecrans, Richard, 55
 Rumsey, Charles B., 31, 90
 Runkel, Jack F., 4, 34, 70, 76
 Runyan, Harry L., 61
 Russell, Walter R., 58
 Rustenburg, John, 29

S

Sadoff, Melvin, 71
 Salters, Leland B., Jr., 94
 Samanich, Nick E., 9, 76
 Savage, Howard F., 22
 Sawyer, Richard H., 89
 Scher, Stanley H., 47

Schmeer, James W., 34, 70, 94
 Schmidt, Stanley F., 71
 Schwind, Richard G., 3
 Seidel, B. S., 48
 Seidel, Barry S., 3
 Senoo, Y., 48
 Serafini, John S., 82
 Sevier, John R., Jr., 10, 30
 Shanks, Robert E., 59, 74
 Short, Barbara J., 77
 Signorelli, Robert A., 53, 82
 Silveira, Milton A., 43
 Sims, Robert L., 41
 Sjoberg, S. A., 58
 Skinn, Donal, 29
 Slye, Robert E., 81
 Smedal, Harald A., 93
 Smith, Charles C., Jr., 15, 30, 39, 44, 74
 Smith, Gerald L., 68
 Smith, Gerald L., 67
 Smith, Willard G., 25
 Sobolev, Yu. S., 83
 Sommer, Robert W., 59
 Sommer, Simon C., 77
 Soulez-Lariviere, Jean, 43, 48, 49
 Spearman, M. Leroy, 64, 68
 Spencer, Bernard, Jr., 40
 Stainback, Calvin, 91
 Stainback, P. Calvin, 23
 Stenning, A. H., 48
 Stephenson, Jack D., 27
 Stewart, Elwood C., 67
 Stickle, Joseph W., 39
 Stitt, Leonard E., 3
 Surina, V. N., 74
 Sutton, Fred B., 74
 Swihart, John M., 4, 28, 76

T

Taillon, Norman V., 17
 Taylor, Lawrence W., Jr., 56
 Taylor, Nancy L., 8
 Taylor, Robert A., 88
 Taylor, Robert T., 19
 Thomas, Andrew G., 5
 Thomas, Lovic P., III, 90
 Thompson, Robert F., 17
 Thompson, William C., 35
 Thomson, Robert G., 6, 86
 Timmons, Jesse D., 47
 Tinling, Bruce E., 61, 65
 Tipps, Daniel O., 29
 Tosti, Louis P., 39, 66, 71
 Tremant, Robert A., 35, 37
 Trussell, Donald H., 6
 Tuovila, Weimer J., 67
 Turner, Thomas R., 16

V

Valensi, Jacques, 43, 48, 49
 Vaughan, Victor L., Jr., 86
 Vedrov, V. S., 74

Vogler, Raymond D., 17
 Vosteen, Louis F., 55

W

Wade, William R., 92
 Walberg, Gerald D., 58
 Waner, Paul G., Jr., 42
 Ward, Robert J., 18
 Ware, George M., 80
 Warner, Paul G., Jr., 49
 Waters, William J., 82, 83
 Watkins, Charles E., 1
 Weeton, John W., 53
 Weiberg, James A., 44
 Weiberg, James A., 10
 Weidman, Deene J., 92
 Welsh, Clement J., 4
 Whitcomb, Richard T., 7
 White, John S., 78
 Whitman, Ruth I., 73
 Whitten, James B., 72
 Wick, Bradford H., 4
 Williams, James L., 65
 Willis, Conrad M., 2
 Wingrove, Rodney C., 29, 93
 Wong, Norman D., 67
 Wong, Robert Y., 52, 55
 Wong, Thomas J., 81
 Woodling, Mildred J., 87
 Woolston, Donald S., 1, 15

Y

Yasaki, Paul T., 51
 Yntema, Robert T., 48
 Yoshikawa, Kenneth K., 4, 18
 Young, George R., 47

REPORT DOCUMENTATION PAGE			Form Approved OMB No. 0704-0188	
Public reporting burden for this collection of information is estimated to average 1 hour per response, including the time for reviewing instructions, searching existing data sources, gathering and maintaining the data needed, and completing and reviewing the collection of information. Send comments regarding this burden estimate or any other aspect of this collection of information, including suggestions for reducing this burden, to Washington Headquarters Services, Directorate for Information Operations and Reports, 1215 Jefferson Davis Highway, Suite 1204, Arlington, VA 22202-4302, and to the Office of Management and Budget, Paperwork Reduction Project (0704-0188), Washington, DC 20503.				
1. AGENCY USE ONLY (Leave blank)		2. REPORT DATE		3. REPORT TYPE AND DATES COVERED
4. TITLE AND SUBTITLE			5. FUNDING NUMBERS	
6. AUTHOR(S)				
7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)			8. PERFORMING ORGANIZATION REPORT NUMBER	
9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES) National Aeronautics and Space Administration Washington, DC 20546-0001			10. SPONSORING/MONITORING AGENCY REPORT NUMBER	
11. SUPPLEMENTARY NOTES				
12a. DISTRIBUTION/AVAILABILITY STATEMENT Subject Category: Availability: NASA CASI (301) 621-0390			12b. DISTRIBUTION CODE	
13. ABSTRACT (Maximum 200 words)				
14. SUBJECT TERMS			15. NUMBER OF PAGES	
			16. PRICE CODE	
17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT	